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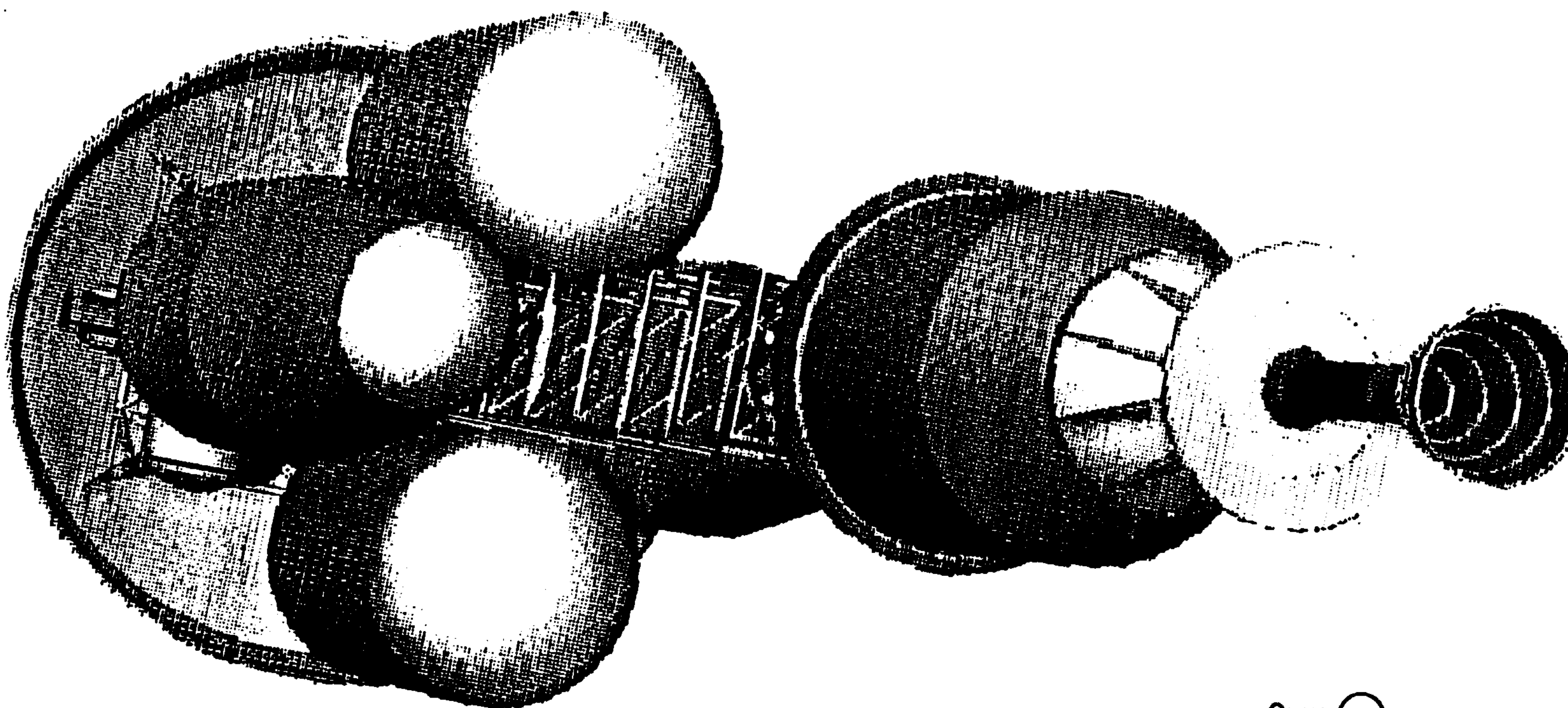

Implementation Plan and Element Description Document (draft final)
Volume 3: Nuclear Thermal RocketVehicle

March 8, 1991

D615-10026-3

ADVANCED CIVIL SPACE SYSTEMS

Boeing Aerospace and Electronics
Huntsville, Alabama

NASA Contract NAS8-37857

528

Space Transfer Concepts and Analyses for Exploration Missions

NASA Contract NAS8-37857

Nuclear Thermal Rocket Implementation Plan and Element Description Document

Boeing Aerospace and Electronics
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Nuclear Thermal Rocket Implementation Plan and Element Description Document

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Documentation Set:
D615-10026-1 IP and ED Volume 1: Major Trades, Books 1and 2
D615-10026-2 IP and ED Volume 2: Cryogenic/ Aerobrake Vehicle
D615-10026-3 IP and ED Volume 3: Nuclear Thermal Rocket Vehicle
D615-10026-4 IP and ED Volume 4: Solar Electric Propulsion Vehicle
D615-10026-5 IP and ED Volume 5: Nuclear Electric Propulsion Vehicle
D615-10026-6 IP and ED Volume 6: Lunar Systems

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Implementation Plan and Element Description Document
Nuclear Thermal Rocket (NTR)
Table of Contents

Symbols, Abbreviations and Acronyms

ACRV	Advanced crew recovery vehicle
ACS	Attitude control system
AFE	Aerobrake Flight Experiment
A&I	Attachment and integration
Al	Aluminum
ALARA	As low as reasonably achievable
ALS	Advanced Launch System
ALSPE	Anomalously large solar proton event
am	Atomic mass (unit)
AR	Area ratio
ARGPER	Argument of perigee
ARS	Atmospheric revitalization system
art-g	Artificial gravity
asc	Ascent
ASE	Advanced space engine
AU	Astronomical Unit (=149.6 million km)
BIT	Built-in test
BITE	Built-in test equipment
BLAP	Boundary Layer Analysis Program
BFO	Blood-forming organs
BMR	Body mounted radiator
C	Degrees Celsius
CAB	Cryogenic/aerobrake
CAD/CAM	Compter-aided design/computer-aided manufacturing
CAP	Cryogenic all-propulsive
C_d	Drag coefficient
CELSS	Closed Environmental Life Support System
CHC	Crew health care
CG	Center of gravity
C_L	Lift coefficient
cm	Centimeter = 0.01 meter
c/m	Crew module
CM	Center of mass
c/o	Check out
C of F	Cost of facilities
conj	Conjunction
COSPAR	Committee on Space Research of the International Council of Scientific Unions
CO2	Carbon dioxide
Cryo	Cryogenic
C3	Hyperbolic excess velocity squared (in km^2/s^2)
C&T	Communications and Telemetry
CTV	Cargo Transport Vehicle (operates in Earth orbit)
d	days
DDT&E	Design, development, testing, and evaluation
DE	Dose equivalent
deg	Degrees
desc	Descent

DMS	Data management system
dV	Velocity change (ΔV)
EA	Earth arrival
E arr	Earth arrival
Ec	Modulus of elasticity in compression
ECCV	Earth crew capture vehicle
ECWS	Element control work station
ECLSS	Environment control and life support system
EP	Electric propulsion
ESA	European Space Agency
e.s.o.	Engine start opportunity
ET	External Tank
ETO	Earth-to-orbit
EVA	Extra-vehicular activity
F_c	Circulation efficiency factor
FD&D	Fire Detection and Differentiation
F_{ew}	Life support weight factor
FEL	First element launch
F_f	Specific floor count factor
F_{fa}	Specific floor area factor
F_i	Aerobrake integration factor
F_l	Specific length factor
F_n	Normalized spatial unit count factor
F_o	Path options factor
F_p	Useful perimeter factor
F_{pc}	Parts count factor
F_{pr}	Proximity convenience factor
F_{rp}	Plan aspect ratio factor
F_{rs}	Section aspect ratio factor
FSE	Flight support equipment
F_s	Vault factor
F_{ss}	Safe-haven split factor
F_u	Spatial unit number factor
F_v	Volume range factor
FY88	Fiscal Year 1988 (=October 1, 1987 to September 30, 1988. Similarly for other years)
g	Acceleration in Earth gravities (=acceleration/9.80665m/s^2)
GCNR	Gas core nuclear rocket
GCR	Galactic cosmic rays
GEO	Geosynchronous Earth Orbit
GN2	Gaseous nitrogen
GN&C	Guidance, navigation, and control
GPS	Global Positioning System
Gy	Gray (SI unit of absorbed radiation energy = 10^4 erg/gm)
hab	Habitation
HD	High Density
HEI	Human Exploration Initiative (obsolete for SEI)
HLLV	Heavy lift launch vehicle
hrs	Hours

hyg w	Hygeine water
HZE	High atomic number and energy particle
H2	Hydrogen
H_2O	Water
ICRP	International Commission on Radiation Protection
IMLEO	Initial mass in low Earth orbit
in.	Inches
inb	Inbound
IP&ED	Implementation Plan and Element Description
IR&D	Independant research and development
Isp	Specific impulse (=thrust/mass flow rate)
ISRU	In-situ resource utilization
JEM	Japan Experiment Module (of SSF)
JSC	Johnson Space Center
k	klb
keV	Thousand electron volt
kg	Kilograms
klb	Kilopounds (thousands of pounds. Conversion to SI units=4448 N/klb)
klbf	Kilopound force
km	Kilometers
KM	Kilometers
KM/Sec	Kilometers per second
KM/SEC	Kilometers per second
ksi	Kilopounds per square inch
LCC	Life cycle cost
L/D	Lift-to-drag ratio
LD	Low density
LDM	Long duration mission
LEO	Low Earth orbit
LET	Linear energy transfer
LEV	Lunar excursion vehicle
LEVCM	Lunar excursion vehicle crew module
Level II	Space Exploration Initiative project office, Johnson Space Center
LH2	Liquid hydrogen
LiOH	Lithium hydroxide
LLO	Low Lunar orbit
LM	Lunar Module
LOR	Lunar orbit rendezvous
LOX	Liquid oxygen
LS	Lunar surface
LTV	Lunar transfer vehicle
LTVCM	Lunar transfer vehicle crew module
L2	Lagrange point 2. A point behind the Moon as seen from the Earth which has the same orbital period as the moon.
m	Meters
[MarsGram	Western Union interplanetary telegram]
[MARSIN	Martian pornography]
MASE	Mission analysis and systems engineering (same as Level II q.v.)
MAV	Mars ascent vehicle

M/C_DA	Ballistic coefficient (mass / drag coefficient times area)
MCRV	Modified crew recovery vehicle
m_e	Mass of electron
MEOP	Maximum expected operating pressure
MeV	Million electron volt
MEV	Mars excursion vehicle
MLI	Multi-layer insulation
mm	Millimeter (=0.001 meter)
MMH	Monomethylhydrazine
MMV	Manned Mars vehicle
MOC	Mars orbit capture
MOI	Mars orbit insertion
mod	Module
M&P	Materials and processes
MPS	Main propulsion system
MR	Mixture ratio
m/sec	Meters per second
MSFC	Marshall Space Flight Center
Msi	Million pounds per square inch
mt	Metric tons (thousands of kilograms)
mT	Metric tons
MTBF	Mean time between failures
MTV	Mars transfer vehicle
MWe	Megawatts electric
m^3	Cubic Meters
N	Newton. Kilogram-meters per second squared
n/a	Not applicable
NASA	National Aeronautics and Space Administration
NCRP	National Council on Radiation Protection
NEP	Nuclear-electric propulsion
NERVA	Nuclear engine for rocket vehicle application
NTP	Nuclear thermal propulsion (same as NTR)
NSO	Nuclear safe orbit
NTR	Nuclear thermal rocket
N2O4	Nitrogen tetroxide
OSE	Orbital support equipment
OTIS	Optimal Trajectories by Implicit Simulation program
outb	Outbound
O2	Oxygen
PBR	Particle bed reactor
Pc	Chamber pressure
PEEK	Polyether-ether ketone
PEGA	Powered Earth gravity assist
P/L	Payload
POTV	Personnel orbital transfer vehicle
pot w	Potable water
PPU	Power processing unit
prop	Propellant
psi	Pounds per square inch
PV	Photovoltaic

Q	Heat flux (Joules per square centimeter)
Q	Radiation quality factor
RAAN	Right ascension of ascending node
RCS	Reaction control system
Re	Reynolds number
RF	Radio frequency
RMLEO	Resupply mass in low Earth orbit
ROI	Return on investment
RPM	Revolutions per minute
RWA	Relative wind angle
R&D	Research and Development
	Rendezvous and dock
SAA	South Atlantic Anomaly
SAIC	Science Applications International Corporation
SEI	Space Exploration Initiative
SEP	Solar-electric propulsion
SI	International system of units (metric system)
SiC	Silicon carbide
SMA	Semimajor axis
sol	Solar day (24.6 hours for Mars)
SPE	Soalr proton events
SRB	Solid Rocket Booster
SSF	Space Station Freedom
SSME	Space Shuttle Main Engine
STCAEM	Space Transfer Concepts and Analysis for Exploration Missions
stg	Stage
surf	Surface
Sv	Sieviert (SI unit of dose equivalent = Gy x Q)
S1	Distance along aerobrake surface forward of the stagnation point
S2	Distance along aerobrake surface aft of the stagnation point
S3	Distance along aerobrake surface starboard of the stagnation point
t.	Metric tons (1000kg)
TBD	To be determined
Tc	Chamber temperature
TCS	Thermal control system
TEI	Trans-Earth injection
TEIS	Trans-Earth injection stage
t.f.	Tank weight factor
THC	Temperature and humidity control
TMI	Trans-Mars injection
TMIS	Trans-Mars injection stage
TPS	Thermal protection system
TT&C	Tracking, telemetry, and control
T/W	Thrust to weight ratio
UN-W/25Re	Uranium nitride - Tungsten/25% Rhenium reactor fuel
VAB	Vehicle Assembly Building
VCS	Vapor coolled shield
Vinf	Velocity at infinity

WBe_2C/B_4C	Tungsten beryllium cabide/Boron cabide composite
WMS	Waste management system
W/O	Without
WP-01	Work package 1 (of SSF)
w/sq cm	Watts per square centimeter (should be Wcm^{-2})
Z	Atomic number
zero g	An unaccelerated frame of reference, free-fall

[order: numbers followed by greek letters]

100K	≤100,000 particles per cubic meter larger than 0.5 micron in diameter
7*n*7	Where n=(0,2-6): Boeing Company jet transport model numbers
°k	Kelvin (K)
+e	Positive charge equal to charge on electron
-e	Charge on electron
ΔV	Change in velocity
s	Standard deviation
μg	Microgravity (also called zero-gravity)

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I. Evolution of Concept

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Concept Development

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EVOLUTION OF THE NUCLEAR THERMAL ROCKET (NTR) VEHICLE

TECHNICAL ARCHITECTURE PRESUMED LEVEL I REQUIREMENTS - During the course of the STCAEM study, and particularly during the *90 Day Study*, many SEI (then HEI) transportation requirements were generated by Office of Exploration Level II. These are reported as appropriate and necessary in various sections of this report, as well as in the *STCAEM Implementation Plan & Element Description Document* technical volumes. Here, space only permits a summary discussion of the Level I requirements adopted by STCAEM as they evolved during the course of the study. The concepts developed and analyzed ultimately were to accommodate the in-space transportation functions required to support the buildup of a permanent presence on the Moon and initial human exploration of Mars. Thus, our Level I requirement was simply to **deliver cargo reliably to the surfaces of the Moon and Mars, and to get people to those places and back safely.** Vehicles in support of missions to *other* destinations are not part of SEI *per se*, and were not addressed by STCAEM. Planet surface system characteristics and Earth-to-orbit (ETO) launch vehicle characteristics were adopted as needed for manifesting purposes, largely intact from other sources. No design work was performed for these two categories. In addition, the mission planning horizon was limited to the year 2025, about 35 years from now.

The chief Level II requirement governing the dimensions of the vehicle concepts we developed came to us during the *90 Day Study*, and was a crew size of 4 for Mars missions. Subsequently, STCAEM performed a simple skill mix analysis or these long-duration missions. Our result was that doubling up on critical skills (for redundancy), given reasonable expectations of how many skills each crew member could become expert in, requires in fact a minimum of 6 - 7 crew members for Mars missions. For the sake of consistency, our vehicle concepts are shown comparable to the *90 Day Study* results, sized for four crew. Impacts accruing from larger crew sizes are discussed in Section x.3.

CONCEPT DEVELOPMENT METHODOLOGY - A vehicle concept emerges gradually through the iterative combination of requirements analysis, subsystems analysis, mass synthesis, performance analysis and configuration design. Because of the cascading, cause-and-effect nature of specific technical decisions in this cyclic process, the ability for a particular concept to remain fully parametric is incrementally lost, sacrificed for depth of detailing. The need to penetrate deeply even at the conceptual stage is twofold: (1) to uncover subtle integration interactions whose ramifications fundamentally revise the concept as they reflect back up the information hierarchy; and (2) to enable the production of graphical images of the concepts capable of being communicated widely *but grounded firmly in engineering detail.* If circumstances allow the concept development process to engage many cycles of reflexive adjustment, from requirements all the way down through subsystem detailing, the design oscillations subside eventually and the product that emerges is a robust and defensible concept. Basic differences in problems posed and solutions engineered lead concept developments in different directions. "Like" problems and solutions gravitate together; their recombination and resolution results in distinct, identifiable vehicle concepts which constitute *vehicle archetypes*. A concept is archetypal if it spawns concept progeny whose ancestry is clear, and if in so doing its salient features recognizably survive subsequent refinement, development and scaling. The ultimate purpose of the STCAEM Concepts and Evolution tasks was to generate, analyze, evaluate and describe such vehicle archetypes, and the role they could play in human space exploration missions.

The STCAEM architecture analysis identified seven major classes of transportation architecture for SEI lunar and Mars missions. Some are derived from different propulsion technology candidates; some are derived from distinct mission philosophies independent of propulsion method; most have many sub-options. Vehicle archetypes are keyed more closely to propulsion method than to mission mode, however, so we found that all seven SEI transportation architectures can be accomplished by derivative combinations of just five archetypal Mars transfer vehicle (MTV) concepts, two archetypal Mars excursion vehicle (MEV) concepts, and one archetypal lunar transportation family (LTF) concept. The concept evolution of these archetypes is outlined in the Major Trades IP&ED book.

DESIGN AND NECKDOWN CRITERIA - STCAEM concept development was punctuated by four "neckdowns", which winnowed down the option candidates generated at each successive level of detail throughout the study. The four neckdowns were intended to result in: (1) **feasible options**, based on promising propulsion technologies capable of performing SEI-class missions; (2) **preferred options**, representing the handful of candidates whose performance and technological readiness were judged to warrant detailed study; (3) **integrated concepts**, vehicle archetypes developed sufficiently to uncover their major integration concerns and architectural context ; and (4) **detailed concepts**, based on the reconciled integration of traded subsystems. The *90 Day Study* occurred such that the first two neckdowns were effectively reversed; cryogenically propelled, aerobraking technology was necessarily preferred at that time, due to depth of understanding. However, STCAEM later rounded out the picture by completing all four neckdown activities, in an ongoing manner throughout the study.

Studying the program architecture implications of various technology options for SEI missions led to the conclusion that the most generally accessible discriminators, *cost* and *risk*, are driven by more subtle technical discriminators than, for instance, initial mass in low Earth orbit (IMLEO). These can be grouped into three broad categories: *feasibility*, *flexibility*, and *multi-use design*. As indicated above, feasibility was the first filter for all concepts considered by STCAEM. **Flexibility** has three components: (1) *robustness*, which is the ability to perform nominally despite variable or unanticipated conditions; (2) *resiliency*, which is the ability to recover from accidental delays or mishaps; and (3) *evolution*, which is an adaptation over time to changing requirements. Flexibility is thus a measure of a program's technical strength and safety in the face of variable extrinsic factors. **Multi-use design** has two components: (1) *re-usability*, which means using the same hardware item more than once; and (2) *commonality*, which means using the same hardware design in more than one setting. Multi-use design is thus a measure of a program's cost-effectiveness and intrinsic longevity. These two key architecture drivers were paramount in interpreting the results of STCAEM's technical trade studies, and figured prominently in the development of element concepts.

MARS TRANSPORTATION - Four Mars transfer propulsion candidates survived all STCAEM neckdowns: cryogenic chemical, nuclear thermal, nuclear electric, and solar electric. Analysis of aerobraking resulted in two performance ranges of interest for Mars entry (hypersonic L/D = 0.5, and L/D = 1.0), as well as the use of high-energy aerobraking (HEAB) for capture at Mars. Consequently, the five archetypal MTV concepts are based respectively on: cryogenic/aerobraking (CAB), cryogenic all-propulsive (CAP), nuclear thermal rocket (NTR), nuclear electric (NEP), and solar electric (SEP) propulsion technologies. The two archetypal MEV concepts are based on the "low" and "high" L/D performance ranges analyzed.

NTR - Nuclear thermal propulsion had a long, successful development history in support of post-Apollo human space exploration , and still occupies a uniquely validated position as a candidate for advanced propulsion for SEI Mars missions. Consequently it was the first advanced propulsion option investigated in depth as a consequence of the *90 Day Study*. Much of the NTR technology discussion centers on sophistication, ranging from resurrections of the original NERVA design, through upgrading that with modern materials, through new particle-bed designs with enhanced I_{sp}, to liquid-core and gas-core rockets. However, this wide range of alternatives is satisfied by one NTR archetype; to first order, only the amount of liquid hydrogen propellant varies among them. The overall vehicle configuration, requirement for shadow shielding of the payload from neutron scattering, long-term storage of LH_2, and solutions for providing artificial gravity, all remain constant. The NTR archetype has existed since the NERVA days; our work has validated it and provided analysis of four particular enhancements: (1) the use of a truss spine instead of a large, structural, axial tank to reduce inert mass; (2) the configuration detailing associated with providing dual engines for engine-out reliability; (3) using a single NTR vehicle to deliver multiple landers to Mars; and (4) truss-spine elongation and careful positioning of large drop-tanks around the mass center to accommodate artificial gravity during all coast phases via end-over-end spinning.

ARTIFICIAL GRAVITY (NTR) - The need for artificial gravity on long-duration interplanetary transfers has not been established. Neither has the *lack* of such a need, however, so STCAEM was obligated to examine the penalties incurred by requiring continuous artificial gravity *en route* between Earth and Mars. Various approaches to rotating artificial gravity have been proposed; STCAEM assessed all of them, and invented some new ones. The fundamental *design* problems associated with artificial gravity derive from: (1) the need for a countermass for rotation; and (2) the high mass cost of precessing the angular momentum vector of a system having large rotational energy. Elegant solutions to both are elusive, and vary widely with propulsion option. Secondary complications are communications and navigation pointing, flight structures sized to hang heavy vehicles, and possibly material fatigue. The fundamental *operations* problems associated with artificial gravity involve crew EVAs during rotation, robotic maintenance in the vehicle's gravity field, crew physiological and psychological responses to a rotating environment, performing minor course-correction propulsive maneuvers and testing the capability prior to departure. Our work has verified that artificial gravity appears feasible for Mars-class missions, for all propulsion options, at fairly modest mass penalties.

The CAP and NTR archetypes accommodate artificial gravity easily. Both are high-thrust systems, so their burn times are extremely short (minutes to hours) compared to coasting transfer time (months). Critical propulsion maneuvers can occur during nonrotating periods of microgravity, at the cost only of spinup/spindown propellant. In general, the propulsion system remaining through the end of the mission can serve as countermass to the contiguously connected habitation systems. When separated by a lightweight truss, they can just spin end-over-end during coast phases to provide sufficient gravity at a comfortable spin rate with acceptable vestibular disturbance (we baselined 1 g to insure full conditioning for surface activity upon arrival at Mars, and 4 rpm maximum spin rate, which together lead to a 56 m separation between the habitat and the center of mass). The additional mass of the truss and propellant for a few budgeted spinup/spindown cycles is of order 10 % of IMLEO.

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Low-L/D Mars Excursion Vehicle (MEV) - The MEV archetype development began during, and was resolved just following, the NASA *90 Day Study*. It was originally conceived as a means of delivering 25 t of undefined payload to the surface of Mars. However, the specification of crew cab provisions, the analysis of vehicle mass balance, and consequently the configuration design of the vehicle all depend on specifics of the payload manifest. We assumed a 20 t reference surface module as an integral part of the MEV. This led to a "Mars campsite" design intended to support a crew of four for 30 - 60 d and became or standard lander design. Chief departures from the lunar campsite mode of operation were:

1) The MEV arrives with the crew already onboard, and so is capable of a really self-contained mission.

2) The MEV also brings with it an ascent vehicle (MAV) with a separate propulsion system, configured optimally for the ascent phase (or ascent after breakaway from the descent stage during a descent abort). The crew cab for the MAV is the operations bridge for the MEV during all its mission phases.

3) The MEV is configured for packaging within an L/D = 0.5 aerobrake. For CAB missions, this brake captures the as-yet unmanned MEV into Mars orbit autonomously, before rendezvous with the MTV, and is used again for the descent. For CAP and other types of missions with propulsive Mars orbit capture, this brake is used only for descent. In all design cases, terminal descent engines are extended through ports in the windward surface of the brake at low Mach number, and the brake is jettisoned subsequently, prior to touchdown.

The MEV configuration was developed to permit later removal and relocation of the surface habitat module, with the aid of surface construction equipment. A variant of the MEV, without either surface module or MAV, was analyzed for delivery of heavy cargo on unmanned missions. A quick assessment was made of the feasibility of re-using an MEV, presuming *in situ* production of oxygen and retention of the aerobrake until touchdown. The outcome was positive, although: (1) additional brake hatches appeared necessary for landing gear deployment, crew egress, and cargo offloading; and (2) a lightweight top-shroud appeared advisable due to aerodynamic drag on ascent, and to permit the crew bridge to protrude beyond the presumed wake-protection limit for direct surface viewing during terminal approach. Configuration options for a "split-stage" MEV, in which the same, or a portion of the same, propulsion system is used for ascent as for terminal descent, were also investigated, and shown to be simple variations of the archetype.

Our baseline aerobrake assembly concept presumed robotic-mediated final assembly of pre-finished, rigid aerobrake segments at *Freedom*. Packaging such segments efficiently by nesting them in an ETO launch shroud is made challenging because of: (1) the aerobrake's asymmetrical, deep-bowl shape, in which the maximum depth of a typical "slice" is comparable to reasonable shroud diameters; and (2) the aerobrake's lip, required for both aerodynamic performance and structural stiffening around the free brake edge. Subsequent manifesting analysis, in which segments were configured according to an initial rib-and-spar structure concept, indicated that two ETO flights would be required to launch a single aerobrake in several pieces. Such extremely volume-limited and volume-inefficient manifesting is an unacceptably poor use of the expensively developed capability that a heavy-lift ETO system represents.

In response to this manifesting problem, STCAEM proposed the "integral launch" concept, in which a fully assembled, integrated aerobrake is launched externally, mounted on the side of the launch vehicle exactly analogous to current STS operations. The low-L/D brake is comparable to the STS orbiter in linear dimensions, and is light enough to launch two at once, with capacity to spare for other, shrouded payload as well. Ascent performance of such a flight configuration requires study; the critical question is whether ascent loads would size the aerobrake structure out of the competitive mass range for the mission itself.

Our structural analysis indicates that since the deep bowl-shaped aerobrake loads like a doubly-curved shell, it may be possible to construct an actual "aeroshell" without resorting to ribs and spars or some other articulated skeletal structure system. The shell would be made of a relatively thin honeycomb-type material system with integral TPS. However, lip buckling would still require a stiff rim, probably facilitated by a closed-tube-section structure. Such a brake may be lighter, and certainly simpler, but the thickened rim would still cause packaging problems due to nesting interference.

High-L/D Reusable Mars Excursion Vehicle (RMEV) - The RMEV archetype development occurred in response to three drivers:

(1) Analysis so far indicates that L/D = 0.5 is sufficient at Mars for controlling an aero-vehicle at Mars. However, the existence of some mission design studies in the literature which advocate L/D > 1.5 for Mars, combined with our preliminary understanding of controllability under Mars conditions, make it important to know in detail how different the configuration constraints imposed by higher L/D would be from those imposed by the lower L/D (which by 1989 had come to be regarded generally as appropriate).

2) As the *90 Day Study* stimulated thinking about what the purpose of SEI Mars surface missions should be, concern developed that global, or at least wide, access to the surface of Mars was potentially important. High-thrust Mars transfer propulsion systems (chemical or NTR) tend to be mass-constrained by arrival and departure vector geometry to certain parking orbit conditions. Although there is no lack of interesting (scientifically important) landing sites accessible from the periapsis of *any* orbit at Mars, the fact that performance-optimized parking orbits are unique for each high-thrust opportunity causes a site-access problem if returning to the same surface site is required (for base buildup). Thus for high-thrust transfer propulsion options particularly, an ability to achieve cross-range on lander entry may be important. High L/D enables greater cross-range capability.

3) Certain Mars lander issues not imposed as requirements during the *90 Day Study* required analysis and design validation. Developing a new MEV concept, substantially different from the baseline MEV, allowed us to investigate those issues simultaneously and thoroughly. Specifically, we addressed: (1) a deep aerobrake structure concept, of interest for maximum structural efficiency and therefore reduced brake mass; (2) the ability to deliver large-envelope cargo manifests, represented in our design by a long-duration surface habitat module sized for 10 crew; and (3) re-usability of the MEV, based on *in situ* production of cryogenic propellant.

The vehicle shape represented by the RMEV has applications for other interesting mission modes, concepts for which have yet to be investigated in detail. Three examples are: (1) a smaller RMEV, sized commensurately with the MEV to be a modest cargo-delivery vehicle; (2) a direct-landing MTV, whose return propellant would be manufactured *in situ* on Mars; and (3) re-usable aerobraked "taxi" vehicles capable of performing the Earth-Mars cycler embark/debark function.

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Nuclear Thermal Rocket (NTR)

Introduction

The reference Mars mission trajectory has a ΔV requirement of 14.6km/s. With chemical propulsion having a Specific Impulse of 475 seconds, the ideal mass ratio for the mission is given by the rocket equation:

$$m(i)/m(f) = e^{(\Delta V/gI(sp))}\text{, in this case } e^{(14600/9.8x475)} = 23 \text{ therefore:}$$
$$m(i)/m(f) = 23:1$$

The start mass, and hence launch cost, savings from reducing the mass ratio can be very large. This is the motivation for considering 'Advanced Propulsion', i.e. propulsion with a higher Specific Impulse than chemical propulsion. Nuclear thermal, nuclear electric, and solar electric are considered in this study, with nuclear thermal considered specifically in this volume.

Nuclear thermal propulsion heats a propellant (usually hydrogen) in thermal contact with the hot core of a reactor. The propellant is heated to about the same temperature as the gas in a chemical rocket (2500-3000K). The much lower molecular weight of hydrogen (2) in a nuclear rocket as compared to water (18) in a chemical rocket leads to a higher exhaust gas velocity, and thus a higher specific impulse (800-1200s vs 450-500s).

Against the much higher propulsion efficiency of nuclear thermal as compared to chemical propulsion, several drawbacks must be considered. Nuclear reactors generate radiation which must be shielded against, especially if it is a manned mission. The engines generally have a lower thrust per unit engine mass. Hydrogen is both extremely cold in liquid form and has a very low density. These require relatively heavy tanks to store it and it is subject to boiloff.

This volume documents analysis and synthesis data developed with respect to nuclear thermal propulsion for a Mars mission. It will be using the same "high-thrust" mission trajectories as the Cryo/Aerobrake vehicle with additional 'fast- trip' trajectories that are now being worked.

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Nuclear Thermal Rocket Vehicle Reference Configuration

Introduction

The nuclear thermal rocket (NTR) concept offers advantages of higher I_{sp} than cryogenic concepts, fully propulsive capture at Mars and Earth to avoid high energy aerobraking, and the potential for recovery and re-use of the expensive transfer habitation system. NTR represents a proven technology; early versions were extensively tested in the 1960s and early 1970s.

Nominal Mission Outline

- The vehicle is assembled, checked out, and boarded in LEO
- The TMI burn occurs, and two empty LH_2 tanks are jettisoned (opposition case)
- The MTV coasts to Mars
- MOI burns capture the MTV into Mars orbit
- Two LH_2 tanks are jettisoned
- The MEV is checked out, separates from the MTV and descends
- The MEV aerobrake is jettisoned prior to final approach
- The MEV touches down, and surface operations ensue
- The MAV ascends for rendezvous with the MTV, leaving the descent stage, surface habitat and science equipment
- The MAV is jettisoned in Mars orbit after crew transfer
- The TEI burn occurs, and the MTV coasts back to Earth
- In expendable scenario, crew return is accomplished with modified ACRV (MCRV), MTV is jettisoned at Earth
- In re-usable scenario, MTV captures propulsively into high parking orbit (500 km by 24 hr) for 30 d cool-down period
- Crew returns to SSF using LEV-class taxi
- Post-cooldown, MTV is refurbished in SSF orbit

Vehicle Systems

Crew Systems

The crew portion of the vehicle consists of a transfer habitat (common with other concepts), deployable PV power plant, and an MEV (common with other concepts). All habitable volumes are contiguously connected, and located at the opposite end of the vehicle from the reactors. The ends of the vehicle are separated by a lightweight truss spine.

Propulsion System

The reactor/engine is a technology-upgrade from the NERVA reactor of the 1970s. A composite shadow shield limits both direct and secondary-particle-scattered dosage to the crew and sensitive electronics. LH_2 propellant is used. Four cryogenic storage drop-tanks are located on the truss. Another, in-line propellant tank is for TEI and EOI; remaining full for most of the mission enables it to provide extra radiation protection to the crew systems. All propellant from the drop-tanks is flowed through the in-line tank, so that its supply remains relatively un-irrad[illegible] throughout the mission.

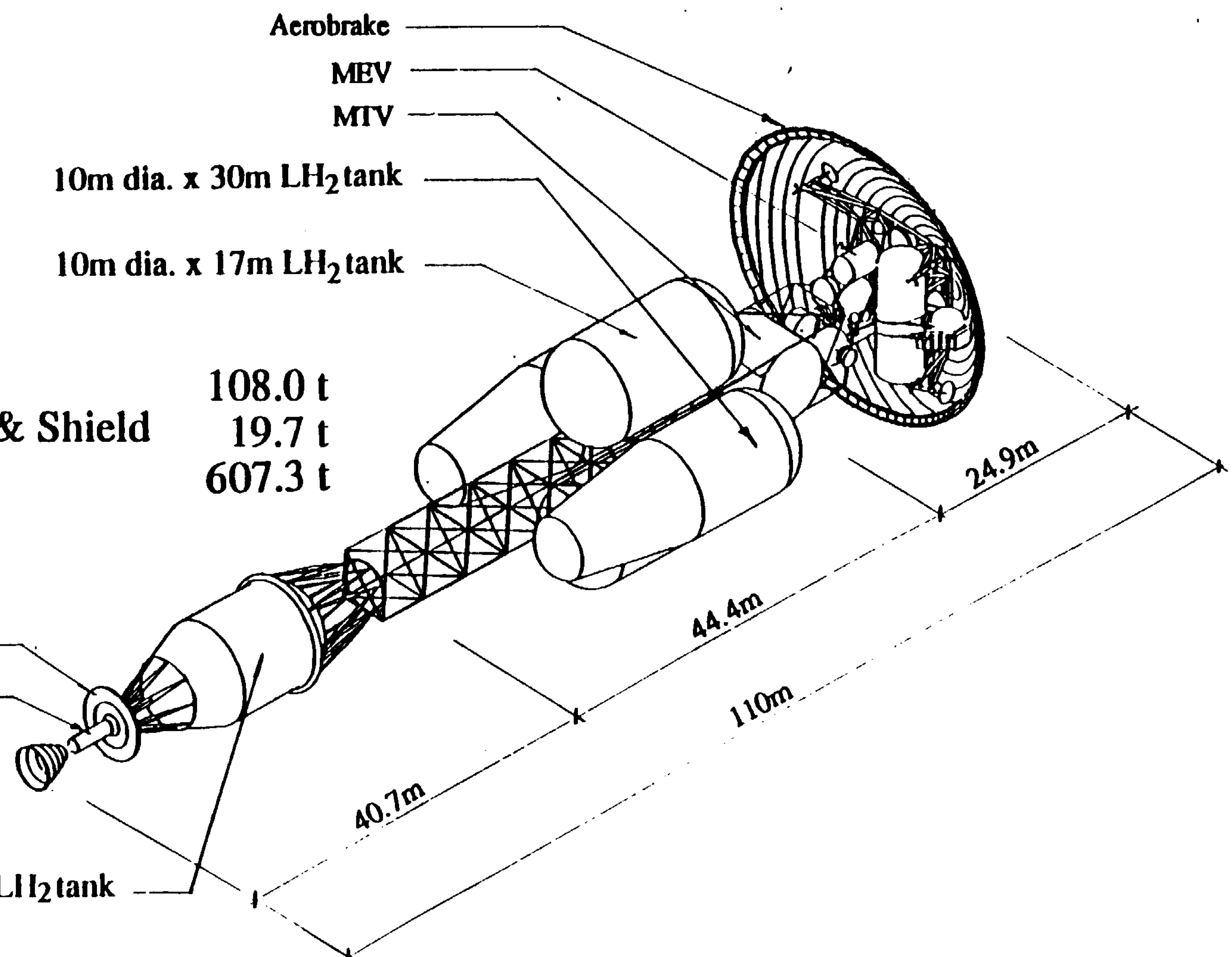
2016 Advanced NERVA NTR Reference Vehicle Configuration

The 925 Isp NERVA derivative engine was chosen by NASA MSFC as the reference propulsion system for the NTR vehicle studies. The performance of the 925 Isp system corresponds to an 'intermediate' reactor fuel element material. Composite fuel elements (see fuel element chart) operating such that the hydrogen propellant reaches approximately 2700°K at 450 psia chamber pressure would provide this Isp, given a large expansion ratio nozzle, and would require no redesign of the NERVA reactor beyond that necessary for integration of these higher temperature fuel elements (cooling and element corrosion are such factors). An Isp of 925 is approximately 85 sec higher than that obtained by the Phoebus 2A reactor in 1967. Such a level of enhancement entails no high risk new technology development, rather it would be an extension of the advanced fuel element analysis that was already underway in the early 1970's when the NERVA program was canceled. Materials development and fabrication techniques in general have seen a lot of advancement in the last 20 years. Thereference vehicle was built around this performance level using the *Boeing Vehicle Synthesis Model*, a sophisticated computer code that outputs vehicle performance figures and weight breakdowns based on very specific vehicle configuration and requirements inputs. The vehicle, as illustrated has four 10 meter dia. hydrogen propellant tanks with a tank fraction of 14%. Two tanks for Earth departure propellant, that are jettisoned after TMI burn, one Mars arrival propellant tank jettisoned after Mars capture and one tank that remains with the vehicle that holds both the Mars departure and Earth arrival propellant. A 2 meter by 35 meter SSF type truss is shown as connecting the in line Mars departure / Earth arrival tank to the 33t, 4 crew habitat module and MEV. The MEV hasa single, low energy, Mars descent only aerobrake - this is not a high energy aerobrake designed for Mars orbit capture. The vehicle does propulsive burns for orbit capture both at Mars and Earth.

NTR Configuration

MEV/MTV	108.0 t
Propulsion, Frame & Shield	19.7 t
Propellant & Tanks	607.3 t

IMLEO 735.0 t



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NTR

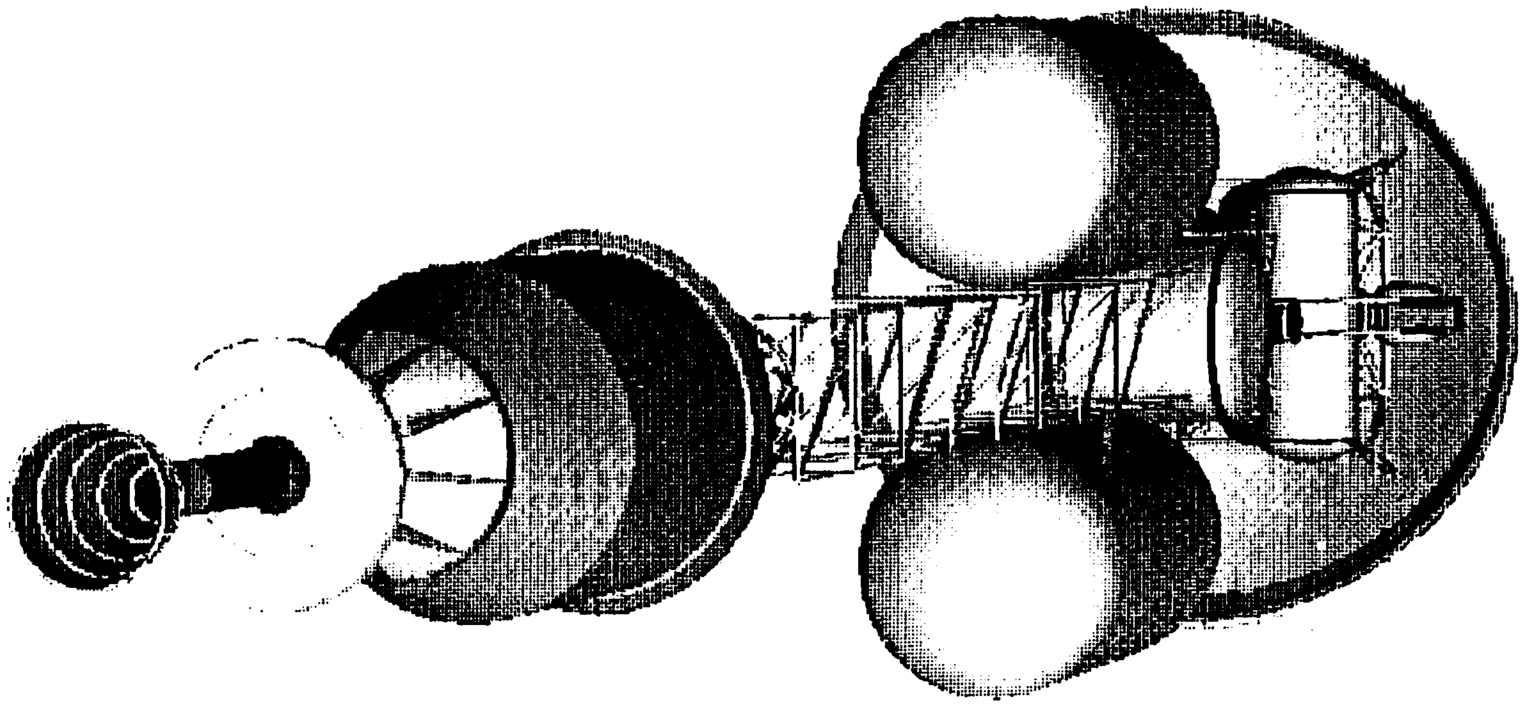
Trades and Rationale

- High Isp
- Fully propulsive capture at Mars and Earth avoids high energy aerocapture.

Mission Modes And Operations

- Vehicle assembled in SSF orbit
- Two LH_2 tanks jettisoned after TMI burn
- Two LH_2 tanks jettisoned after MOI burn
- MEV/Aerobrake separate from vehicle prior to entry and landing
- Aerobrake separates from MEV prior to landing.
- Crew cab ascent after surface mission, leaving lander, surface hab.
- Crew cab left in Mars orbit after rendevous, docking and crew transfer.
- TEI burn
- EOI burn and crew return to SSF.

Reference Configuration: *Outbound and Mars Capture*



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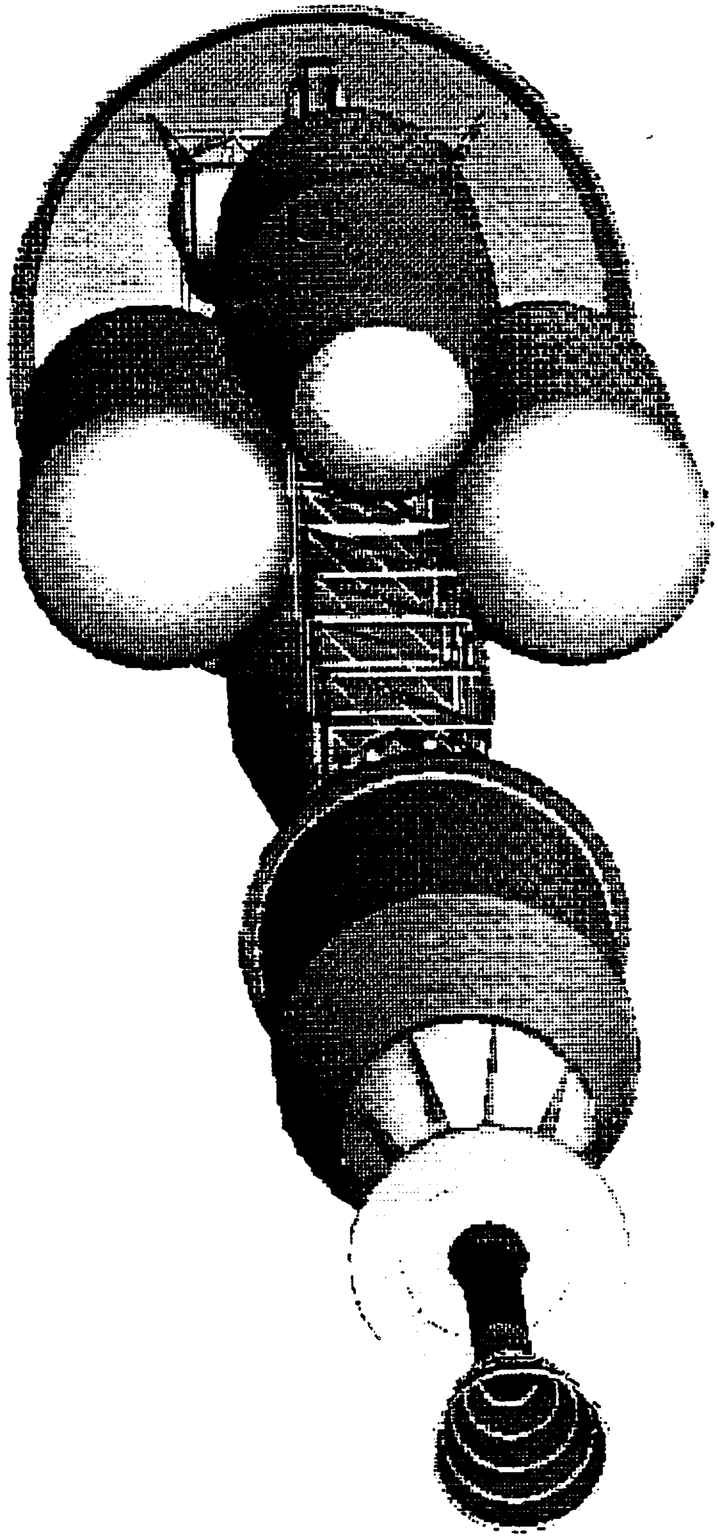
Reference Configuration: *Earth Departure*



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Options / Alternatives

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NTR Alternate Vehicle Configurations

The Nuclear Thermal Rocket is not a new idea, as it stated elsewhere in this text it has a history of hardware and ground tests. Although a flight ready vehicle was never produced, it was designed, giving us a starting point for our evaluations. In modernizing the designs the structure of the vehicle, particularly the truss and tank systems were adapted to the available or proposed ETO vehicle shroud size and throw-weight. In addition the alternative of incorporating the tanks as part of the structure of the vehicle was traded. In this version of the vehicle the tanks were not expendable (no lightening of load for burn maneuvers subsequent to TMI) and the mass penalty incurred in strengthening the tanks to take the load did not seem to make this an attractive option. One other factor was incorporated in the design consideration at this point. That was the results of the SAIC radiation studies evaluating the shield size needed and the view factor of the reactor. This resulted in reshaping the reactor side portion of the fuel tanks.

NTR VEHICLE CONFIGURATION OPTIONS

The nuclear engine greatly influences the overall physical configuration of any NTR vehicle. The necessity for radiation attenuation between the engine source and the crew as well as the placement and staging of very large hydrogen propellant tanks are two major considerations that are unique to NTR systems. The following factors are applicable in this regard:

(1) Radiation dosage received by crew = 1/(separation distance) squared

Separation distance between the crew and reactor is a key player in reducing the amount of reactor generated radiation that reaches the crew habitat module. Since the reactor radiation dosage that eventually reaches the hab module is equal to the inverse of the separation distance squared, grouping the lengthy propellant tanks into a axial alignment rather than a radial cluster maximizes radiation attenuation by maximizing the separation distance provided by the tankage/structure without unduly penalizing the vehicle with structure dedicated solely to extending separation distance. Doubling the separation distance reduces the received dosage by a factor of 4.

(2) Axial alignment of tanks rather than radial clustering also allows the reactor radiation shadow shield protected cone half angle to be smaller since there would be less projected tank area around the reactor that could scatter direct radiation and thus become a secondary source. Any reactor shadow shield would include a very dense layer of material such as tungsten dedicated solely to gamma ray attenuation. Minimizing the shield size is important in keeping the weight down.

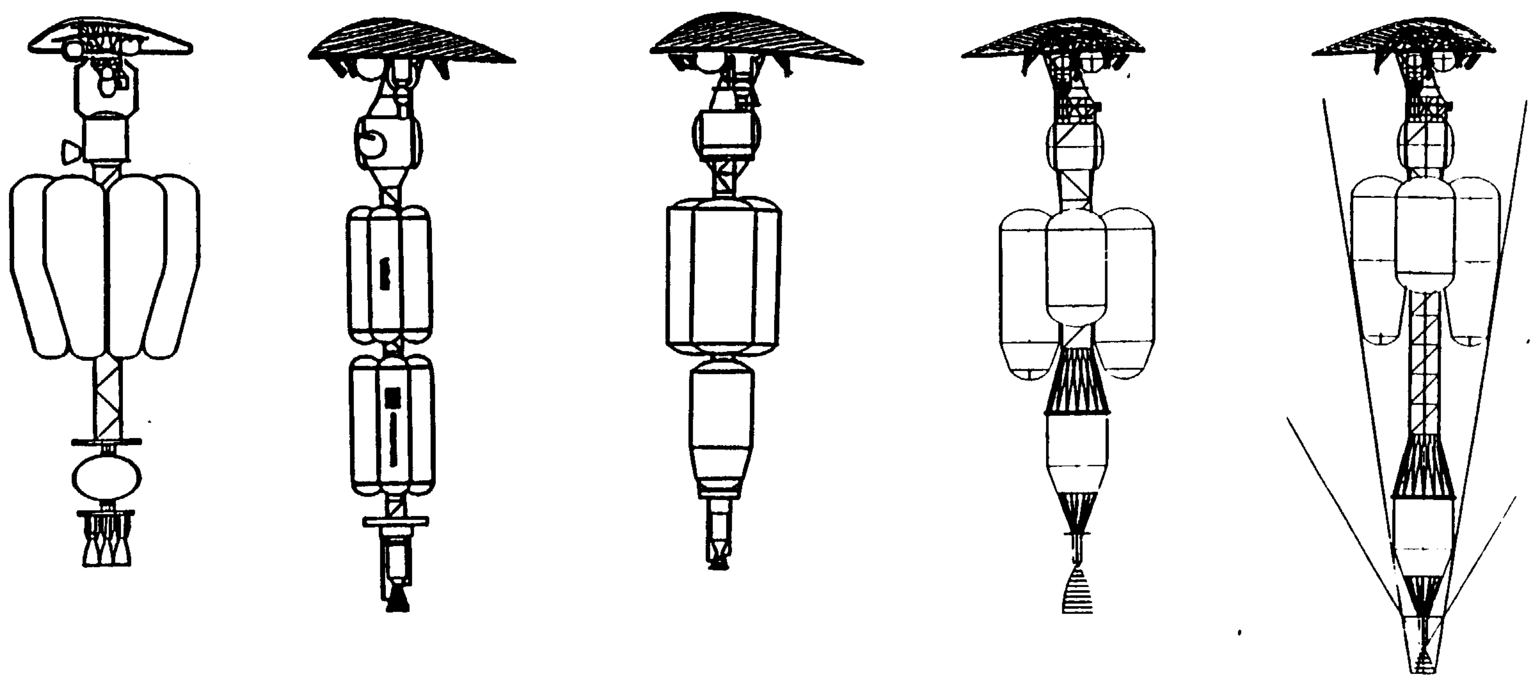
(3) Axial alignment provides more hydrogen propellant to be utilized as a secondary thermal neutron shield in the direct line between the crew cab and the reactor.

The configurations shown below are representations of various tank size and tank placement options. It is beneficial from a shielding viewpoint to keep the Earth arrival propellant in an 'inline' tank just behind the reactor shield. It is beneficial from an IMLEO weight standpoint to:

(a) jettison the tanks after each burn
(b) use as large a tank size as the launch vehicle(s) can deliver
(c) use advanced materials such as metal matrix composites to keep the tank fraction as low as possible

Other Issues include: Providing for tank release and jettison, minimizing and facilitating on orbit assembly, anticipating meteor shielding requirements (with or without a protection hanger at SSF), vehicle return for reuse refurbishment/resupply issues, artificial g accommodations and others.

2015 Reference NTR Design History



Earliest concept	Shuttle-C Tanks	10 meter dia tanks	CAD-CAM 5 tanks	SAIC Radiation assessment

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Mars Flyby with Surface Exploration Mission (dash/flyby)

The Mars flyby with surface exploration mission, or 'dash/flyby' mission, is a special mission case that provides modest total mission dVs, with intermediate trip times (400-550).

A Mars flyby trajectory is shown in the following sketch with a dash off the trajectory path occuring past the midpoint of the outbound transfer leg. The dashed line extending out to Mars ahead of the main flyby path illustrates the accelerated MEV flight portion that is a distinctive characteristic of the dash/flyby mission.

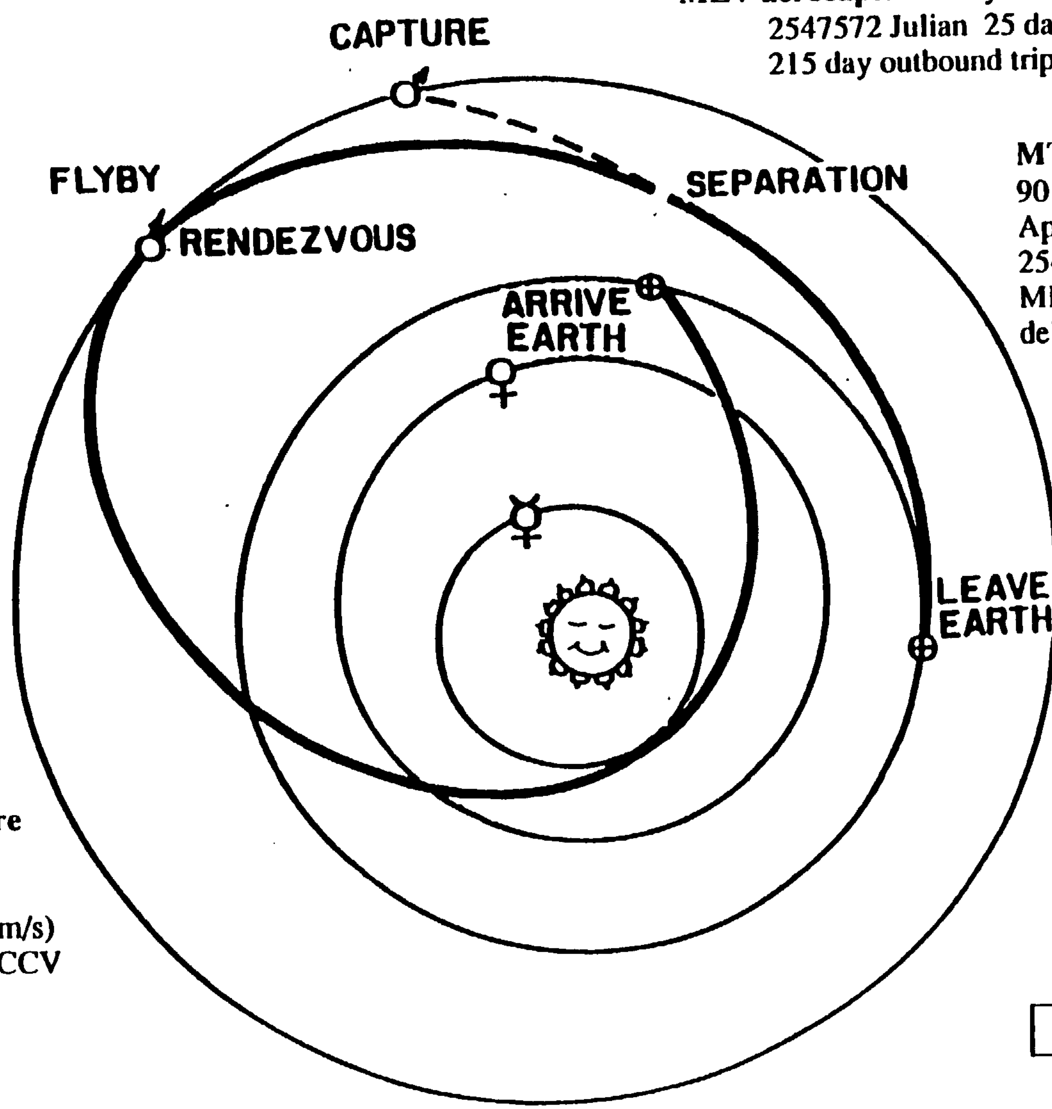
ADVANCED CIVIL SPACE SYSTEMS

2016 Mars Flyby Trajectory with Venus Swingby and Manned Surface Mission

BOEING

MEV aerocapture July 2 2016
2547572 Julian 25 day stay time
215 day outbound trip time

MTV powered Mars swingby
August 2 2016
2547602 Julian
delta V = 806 (km/s)



MTV-MEV Separation
90 days before MEV arrivial
April 26 2016
2547487 Julian
MEV kick stg
delta V = 500 (km/s)

MTV unpowered Venus swingby
March 4 2017
2547816 Julian

Earth Departure
Dec 6 2015
2547362 Julian
TMI delta V ideal=
4271 (km/s)

ECCV Earth capture
April 18 2017
2547861 Julian
Earth Vhp = 10641 (m/s)
slowed to 9700 by ECCV
propul braking stage

Total trip time = 499 days

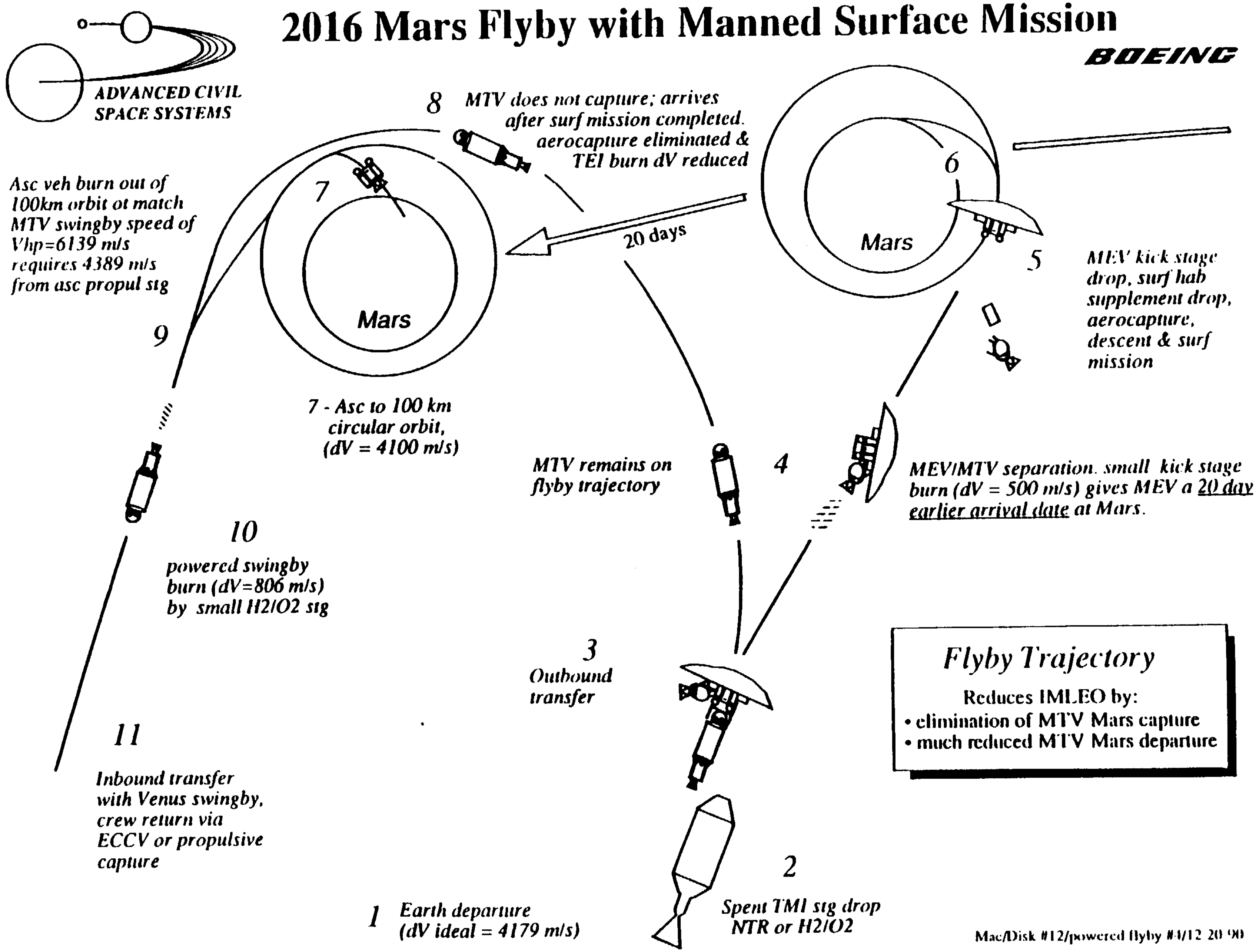
/STCAEM/bd/11Jan91

Flyby/'Dash' Mission profile:

The spacecraft departs Earth on a Mars flyby trajectory and continues on until past the midpoint of the outbound leg. At a specified time the MEV will separate from the main MTV stage and with a small chemical kick stage do a deep space burn of relatively low dV (~500 m/s). The burn provides the MEV with enough extra velocity for it to reach Mars 20+ days ahead of the MTV stage which has continued on its original trajectory. The MEVs new accelerated transfer leg (shown as the dashed line) varies in duration, depending on kick stage dV and the time of separation from the MTV. An extra 4 ton module, supplementing the MEV surface habitat module utilized by the MEV crew for this leg is dropped just prior to aerocapture.

After Mars orbit capture, the MEV does a traditional surface mission, culminating with crew ascent to a waiting parking orbit via the ascent vehicle. The incoming MTV stage does not capture, but rather swings by Mars in a hyperbolic turn. The ascent vehicle does a second burn to depart orbit and establish a hyperbolic path identical to, but ahead of the the MTV path, then effects a rendezvous when the MTV catches up several Mars radi distant from the planet. After crew transfer, the MTV stage performs a modest powered swingby burn. An unpowered Venus swingby occurs on the inbound leg.

2016 Mars Flyby with Manned Surface Mission



Flyby Trajectory

Reduces IMLEO by:

- elimination of MTV Mars capture
- much reduced MTV Mars departure

Mac/Disk #12/powered flyby #4/12 20 90

/STCAEM/bd/11Jan91

Mars Flyby/'Dash' mission

Surface Crew Rendezvous with MTV on Swingby

After Mars orbit capture, the MEV does a 20 to 50 day surface mission, culminating with crew ascent to a waiting parking orbit via the ascent vehicle. The incoming MTV stage does not capture, but rather swings by Mars in a hyperbolic turn. The following chart illustrates the two propulsion options available to the ascent vehicle to do the departure and rendezvous maneuver.

Option 1: Ascent vehicle departure burn from parking orbit

After ascent to the Mars parking orbit, the ascent vehicle does a second burn utilizing its own engines and propellant to depart orbit and establish a hyperbolic path identical to, but ahead of the the MTV path. It then effects a rendezvous when the MTV catches up to it several Mars radii distant from the planet. After crew transfer, the MTV stage performs a modest powered swingby burn.

Option 2: Kick stage (linked to spent ascent stg) departure burn from orbit

After ascent to the Mars parking orbit, the ascent vehicle rendezvous with the small kick stage. The kick stage was first utilized for the deep space MEV kick injection burn for early Mars arrival. Its second task (option 2 only) is to do the orbit departure burn for rendezvous with the MTV stage. The MEV descent stage, ascent stage and this small chemical kick stage are aerocaptured into orbit behind the MEV aeroshell. Before descent, the kick stage is released and stays in orbit while the MEV descends to the surface. Once the ascent vehicle rendezvous with the kick stage, the kick stage does the departure burn to establish a hyperbolic path identical to, but ahead of the the MTV path. Rendezvous occurs, when the MTV catches up to it several Mars radii distant from the planet. After crew transfer, the MTV stage performs a modest powered swingby burn. This approach its similar to the LEM/service module relationship used on the Apollo missions except in this case the small kick stage is unmanned, and is expended after the crew transfers from the ascent/kick stage combination to the MTV stage.

The advantage of option 2

is that the propellant necessary to leave Mars orbit and to match the hyperbolic velocity of the MTV on swingby, does not have to be taken down to the surface and back up to orbit again, as it would if the ascent stage only (option 1) did the orbit dep burn on its own. This saves total MEV system weight, though for some of the Flyby/dash missions investigated, the MTV swingby Vhp was low enough that the penalty of added propellant necessary for the asc vehicle to do the dep burn was not enough to warrant the use of this kick stage, 2020 was such a mission, and option 2 was not utilized for those analysis.

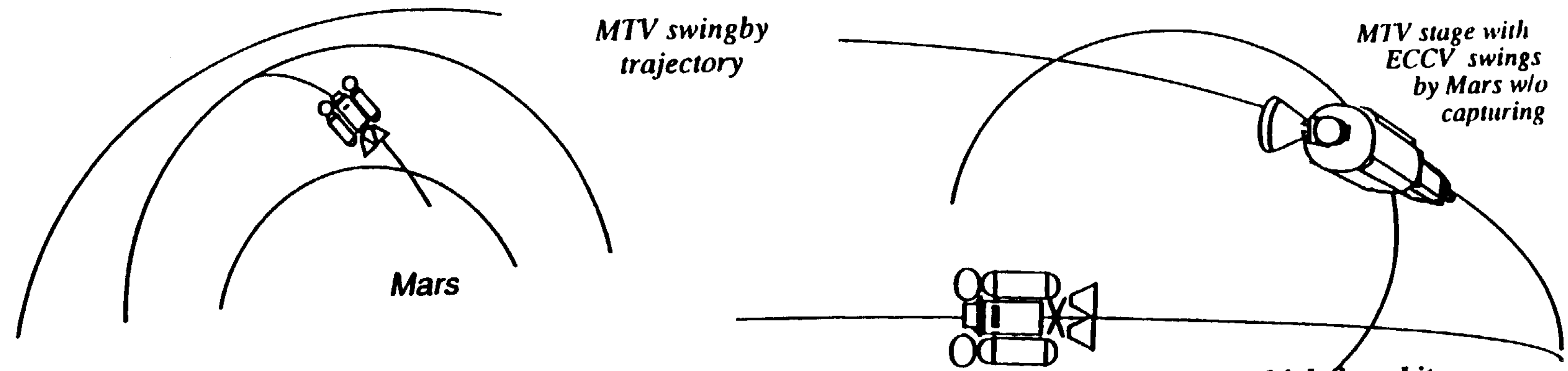
ADVANCED CIVIL SPACE SYSTEMS

Mars Flyby with Surface Exploration Mission

Surface Crew Rendezvous with MTV on Swingby

BOEING

Option 1: Ascent Vehicle does hyperbolic departure from Mars orbit

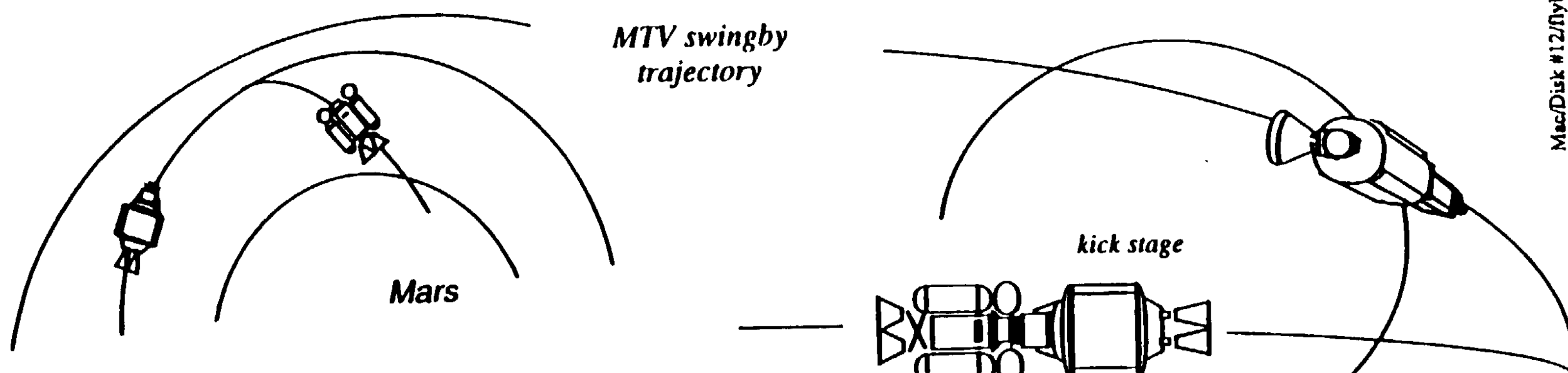


1 ***Ascent from surface***
first burn to orbit; propellant tank drop

2 ***Ascent vehicle in orbit***
awaiting arrival of MTV

Option 2: Kick Stage (with spent asc stg) does hyperbolic departure from Mars parking orbit

decreases MEV asc stg prop load compared to option 1 above; trade off lowers total IMLEO



1 ***Ascent from surface***
kick stage in orbit

2 ***rendezvous with kick stage in orbit***
awaiting swingby of MTV

Mac/Disk #12/flyby
traj asc to rend/1-2-90

/STCAEM/bd/11Jan91

Mars Flyby/'Dash' mission

MEV 'DASH': Deep Space Injection dV Addition

Some where after midpoint in the Mars vehicle outbound transfer leg, the MEV must be accelerated to a higher velocity to allow it to arrive 20 - 50 days ahead of the MTV for surface exploration. This dV addition can be done two ways. The following chart illustrates the two propulsion options available to the MTV/MEV vehicle to do the 'DASH' early Mars deep space injection burn (kick burn) maneuver:

Option 1: Separate chemical kick stage burn for MEV dV addition

Left hand figure: the MEV separates from the MTV stage and its kick stage does the burn. The kick stage is either dropped before MEV aerocapture or retained for further use as a Mars depature kick stg for the ascent vehicle. see 'Surface Crew Rendezvous with MTV' chart.

Option 2: Main propulsion stage gives added velocity.

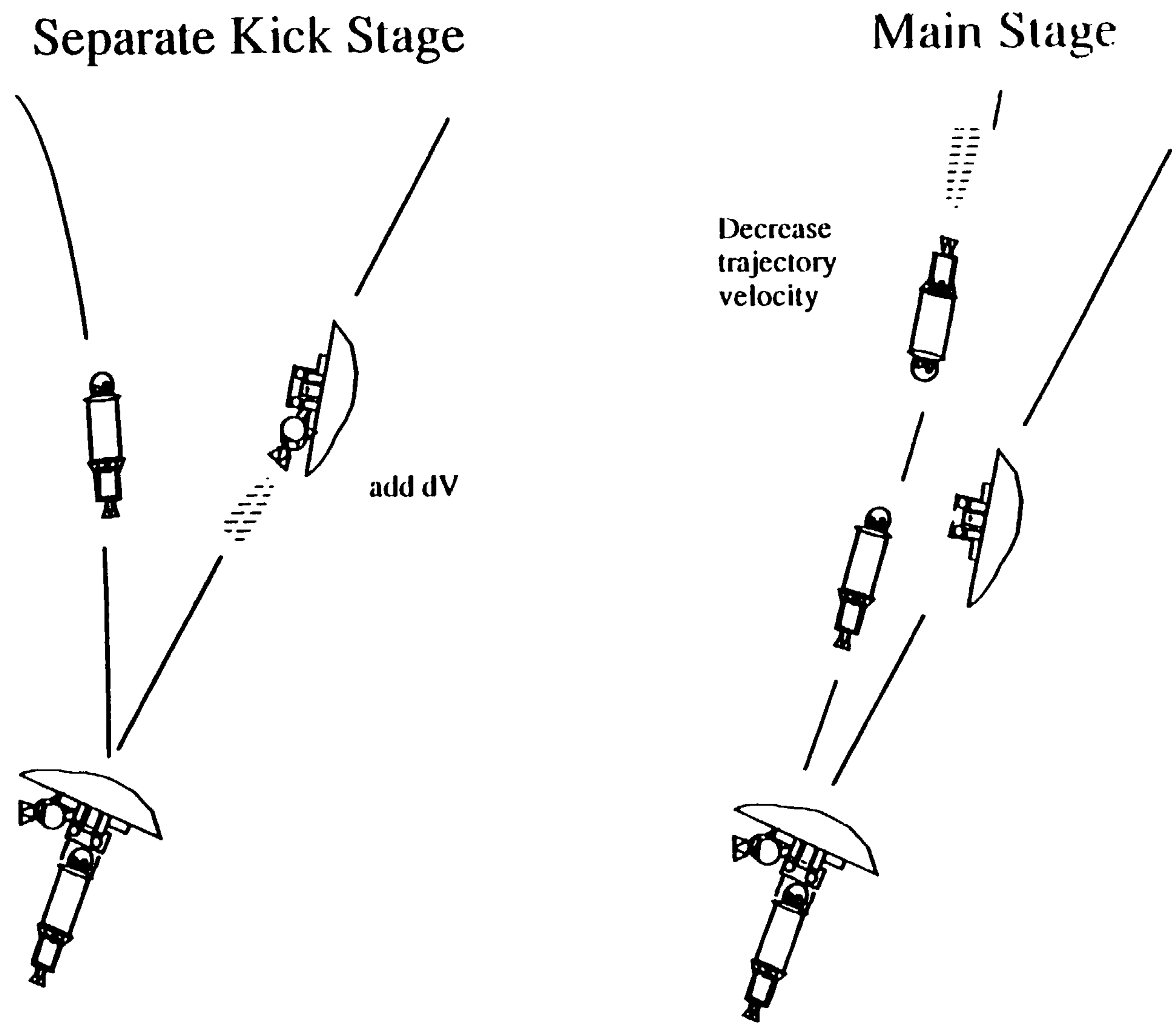
Right hand figure: The vehicle TMI stage leaves 20-50 days earlier than option one, and at the appropriate time in the outbound transfer leg the MEV and MTV stages separate. At that point the MTV stage realigns itself with the engines in the direction of the flight path and the MTV does a small burn to *decelerate* so that it arrives at Mars (for swingby, not capture) 20 - 50 days *after* the MEV.

Advantages of Option 2:

1. No need for a separate small MEV kick stage - one less item to develop

2. Allow the vehicle to take advantage of the higher Isp main stage for this burn. for the NTR vehicles the difference is 925-1050 sec vs 475 sec chemical - lowers IMLEO

MEV 'Dash': Deep Space Injection dV



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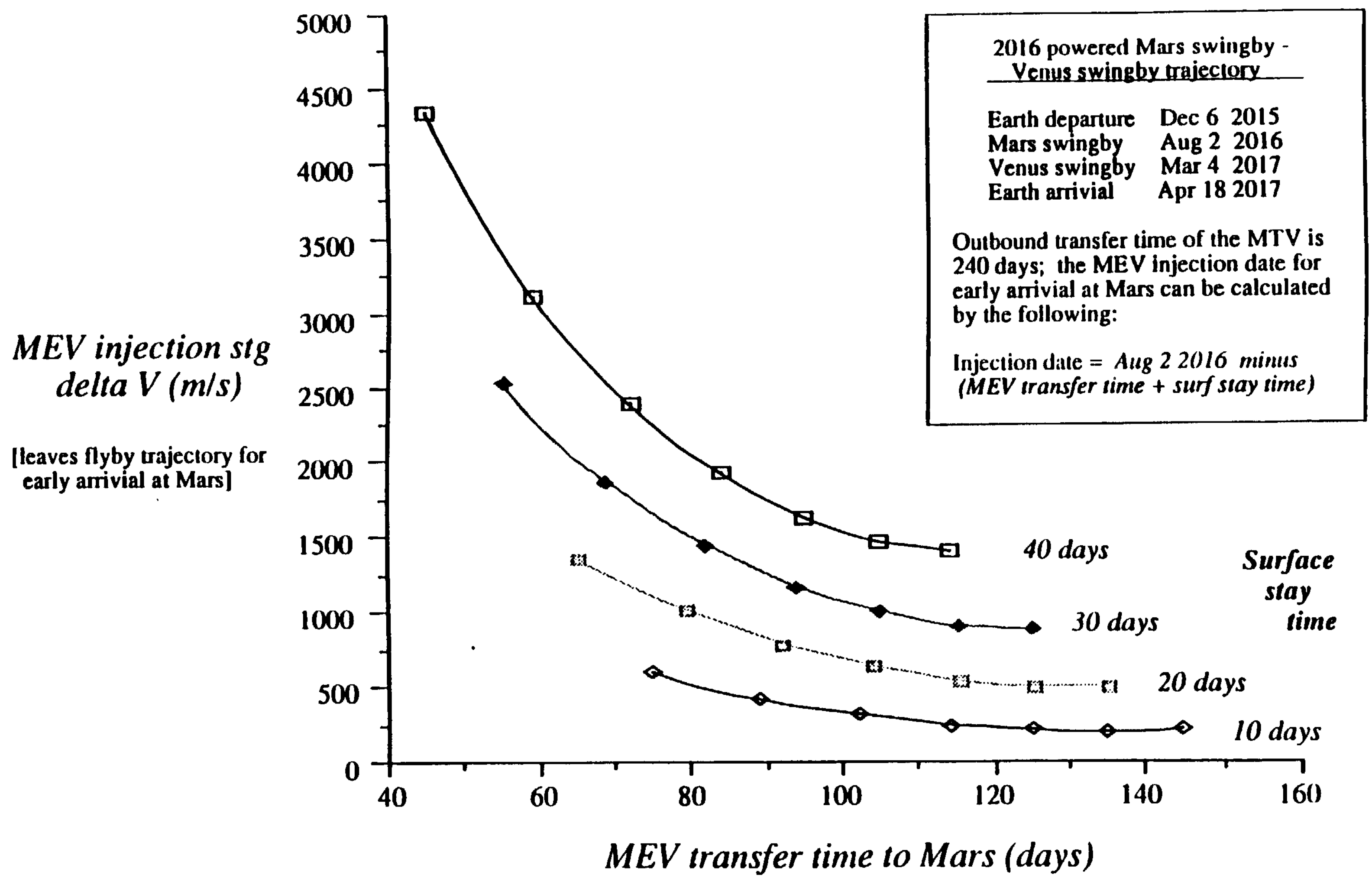
MEV Kick Stage dV vs Surface Stay Time and Transfer Time

The accelerated MEV flight portion is a distinctive characteristic of the dash/flyby mission trajectory. The calculation of the dV required of the kick stage for a given surface stay time is given in the following two charts.

2016 MEV Injection dV vs Surf Stay & Transfer Time

dV required for MEV to reach Mars early enough to perform surface mission, then ascend to rendezvous with MTV as it swings by planet



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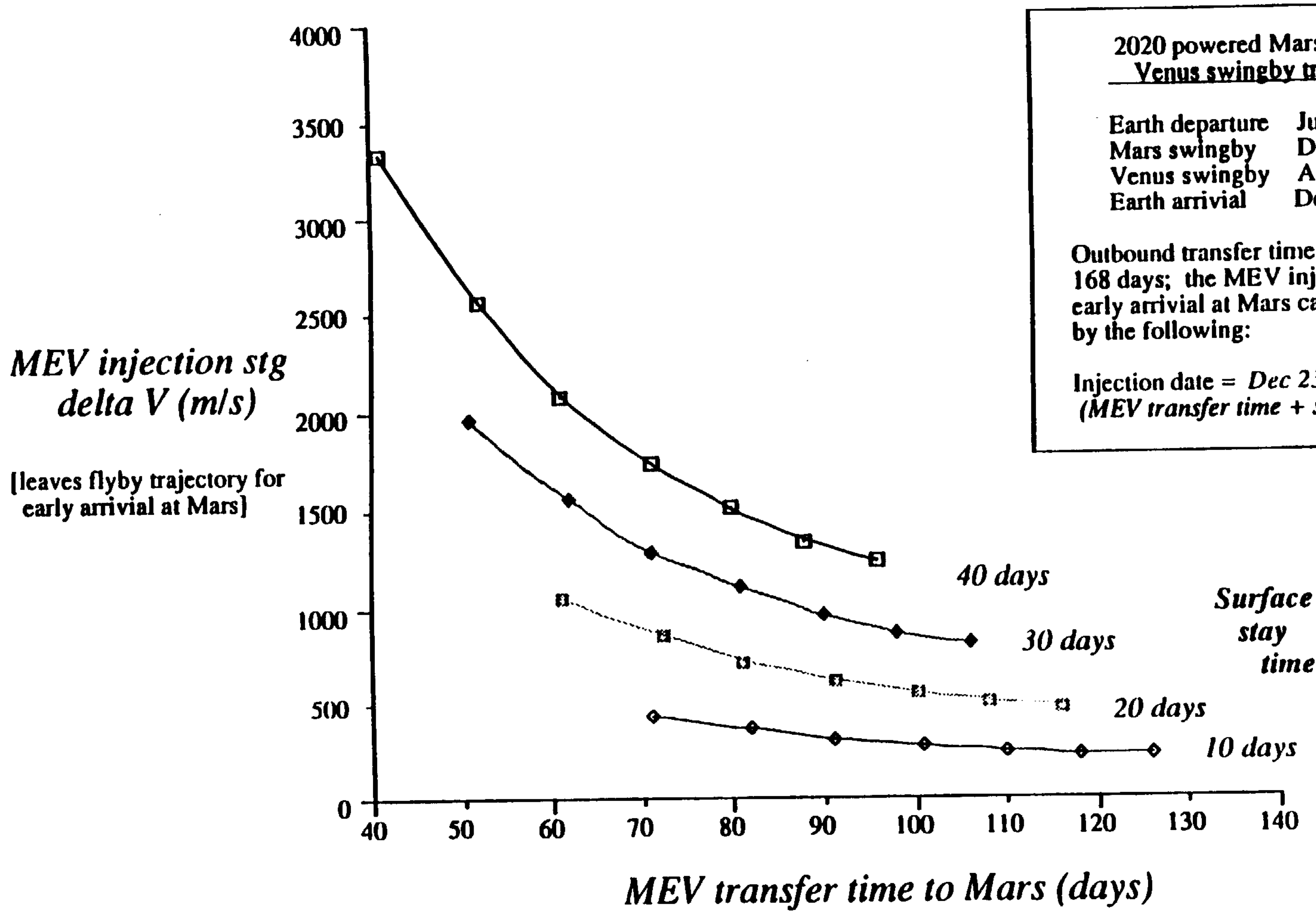
2020 MEV Injection dV vs Surf Stay & Transfer Time

dV required for MEV to reach Mars early enough to perform surface mission, then ascend to rendezvous with MTV as it swings by planet



2020 powered Mars swingby - Venus swingby trajectory

Earth departure	July 8 2020
Mars swingby	Dec 23 2020
Venus swingby	Aug 3 2021
Earth arrivial	Dec 17 2021

Outbound transfer time of the MTV is 168 days; the MEV injection date for early arrivial at Mars can be calculated by the following:

Injection date = *Dec 23 2020 minus (MEV transfer time + surf stay time)*

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Total Vehicle Momentum Change Comparison

The advantage of the Flyby/'Dash' mission approach lies in the way the total mission dV is distributed among the MTV and MEV stages. Two major velocity changes at the target planet are eliminated for the heavy MTV stage. Mars orbit capture and Mars orbit departure burns are now unnecessary, and only the modest powered swingby burn (800 - 1600 m/s dV) is required. The large Mars departure burn is now only assessed against the very light ascent stage, which must not only achieve orbital velocity, but escape velocity as well.

Eliminating the MTV stage MOC requirement altogether, and redistributing the TEI dV to the ascent stage is used as a means of redueing overall vehicle IMLEO.

Savings are realized because the total cumulative MTV and MEV momentum change is significantly reduced. Momentum change is a better indicator than dV of the total mission impulsive energy required of the propulsion system. For the dash/flyby type missions, the total dV required of the ascent stage is increased, but its low inert mass effects only a modest overall system momentum increase, and thus only a small overall vehicle system mass increase. The large MOC and TEI dVs of the stopover trajectories, multiplied by the heavy MTV stage mass they are applied to, results in a large total momentum change; with a corresponding large weight penalty.

By eliminating the MOC dV and reducing the dV required leaving Mars (from a TEI orbit departure burn to a powered swingby burn), the necessary MTV stage momentum change for the flyby case is reduced by an order of magnitude compared to the stopover mission, and is the key advantage offered by this mission type. In order to illustrate the importance of the dV distribution among the stages, a momentum account profile showing five major momentum changes, encountered during the mission, is given below.

ADVANCED CIVIL SPACE SYSTEMS

Vehicle Total Momentum Change Comparison

BOEING

Comparison of Momentum Changes
2016 'dash/flyby' and 'stopover' opposition missions
NERVA derivative NTR propulsion, expendable mode

Total mission momentum = maneuver dV (velocity) times maneuver payload mass for all maneuvers

stage	TMI dV (m/s)	TMI payload (tons)	Kick stg dV (m/s)	Kick stg p/l (t)	MOC dV (m/s)	MOC payload (tons)	Descent dV (m/s)	Descent p/l (t)	Ascent dV (m/s)	Ascent p/l (t)	TEI dV (m/s)	TEI p/l (t)		Total Mission Momentum Change (kg-m/s) x 1,000,000	Vehicle IMLEO (tons)
Stopover Opposition:															
MTV	3851	311 t			3870	175 t					3900	63 t		2.12	
MEV			n/a	n/a			4200	53 t	4100	7 t				0.25	
													total=	2.37	565
Dash/flyby Opposition:															
MTV	4271	199 t			0	62 t					806	62 t		0.90	
MEV			**500	94 t			4200	58 t	8474	10 t				0.38	
													total=	1.28	374

** O2/H2 chem kick stage utilized to accelerate MEV and 4 ton supplemental hab to early Mars arrival for surface mission

Mac/Disk #13/comparison of momentum/1-9-91

Mars Flyby/'Dash' mission

2016 IMLEO vs MEV Wt & Injection dV for Early Mars Arrival

Vehicle IMLEOs were calculated for the Mars Flyby/Dash mode to determine the savings in vehicle weight over the traditional stopover opposition missions. Four vehicle propulsion types were analyzed:

MEV	TMI stage	powered swingby stage	column
1.	Chemical O_2/H_2	Chemical O_2/H_2	left
2.	NEVA NTR	Chemical O_2/H_2	right

MEV weight was varied to show IMLEO delta's to the MEV cargo delivered (to Mars surface) weight. For this chart Ascent departure from Mars orbit to rendezvous *Option 1* was utilized.

Option 1: Ascent vehicle departure burn from parking orbit

After ascent to the Mars parking orbit, the ascent vehicle does a second burn utilizing its own engines and added propellant to depart orbit and establish a hyperbolic path identical to, but ahead of the the MTV path. It then effects a rendezvous when the MTV catches up to it several Mars radii distant from the planet. After crew transfer, the MTV stage performs a modest powered swingby burn.

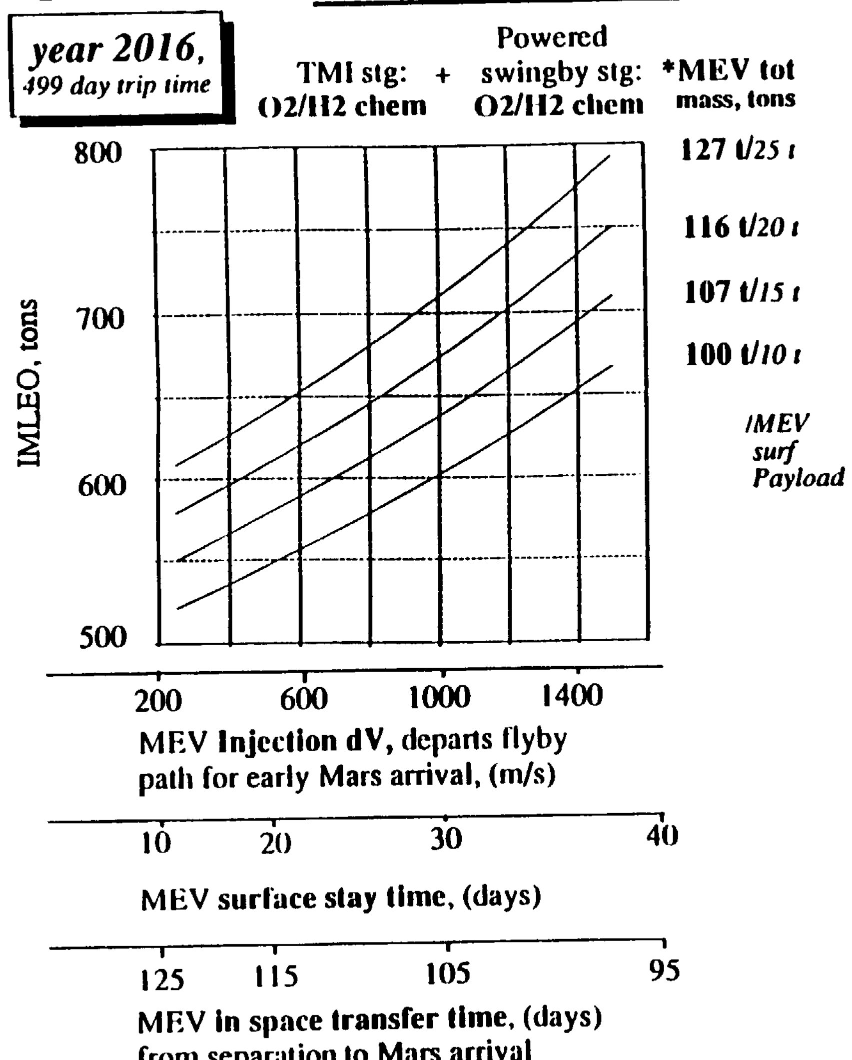
A similar IMLEO vs MEV st chart follows the one below to show weight saving afforded by using option 2 - kick stage departure.

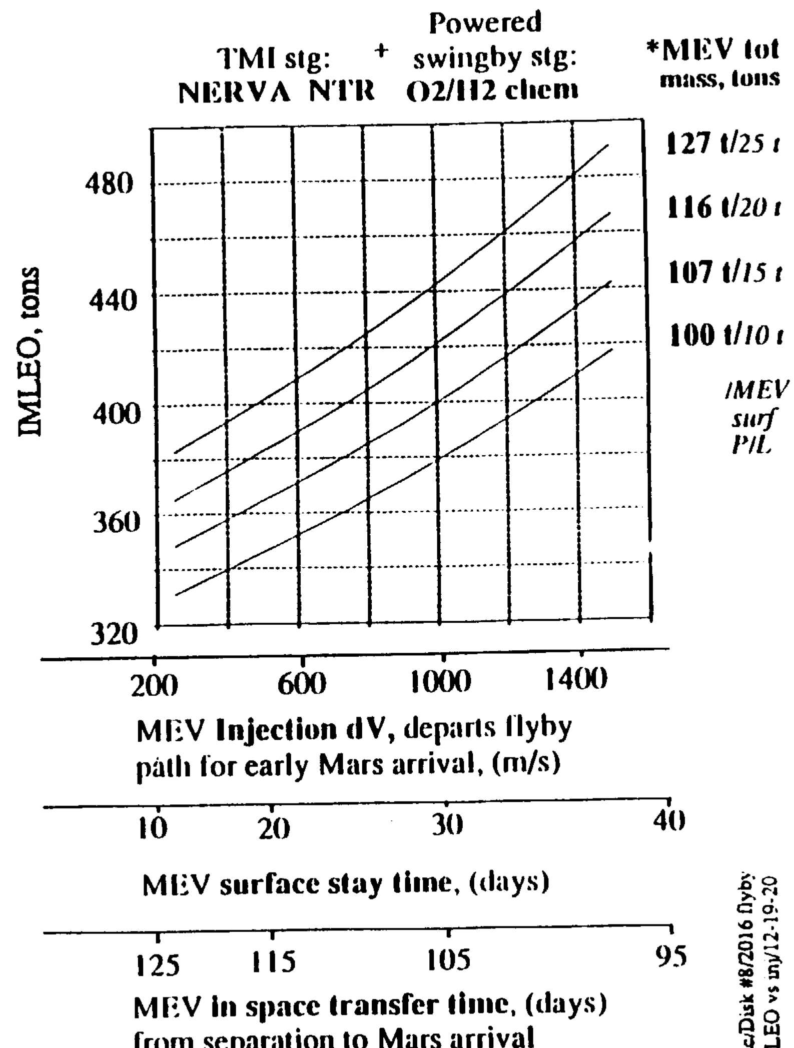
Mars Flyby Missions with Surface Exploration:

IMLEO vs MEV Wt & Injection dV for Early Arrival

Option 1: Ascent vehicle direct rendezvous w MTV on hyperbolic swingby path

ADVANCED CIVIL SPACE SYSTEMS — BOEING





* *Asc veh ascends to rendezvous w MTV on hyperbolic swingby path, thus MEV wt is function of swingby Vhp*

Mac/Disk #8/2016 flyby
IMLEO vs mj/12-19-20

/STCAEM/bd/11Jan91

Mars Flyby/'Dash' mission

2016 IMLEO vs MEV Wt & Injection dV for Early Arrival

Vehicle IMLEOs were calculated for the Mars Flyby/Dash mode to determine the savings in vehicle weight over the traditional stopover opposition missions. Four vehicle propulsion types were analyzed:

	TMI stage	powered swingby stage	column
1.	Chemical O_2/H_2	Chemical O_2/H_2	left
2.	NEVA NTR	NERVA NTR	right

MEV weight was varied to show IMLEO delta's to the MEV cargo delivered (to Mars surface) weight. For this chart *Option 2* was utilized.

Option 2: Kick stage (linked to spent ascent stg) departure burn from orbit

After ascent vehicle rendezvous in orbit, the small kick stage does the departure burn to establish a hyperbolic path identical to, but ahead of the the MTV path. Rendezvous occurs, when the MTV catches up to it several Mars radii distant from the planet. This saves total MEV system weight for the 2016 mission as can be seen from a comparison of the previous IMLEO chart of Option 1.

Mars Flyby Missions with Surface Exploration:

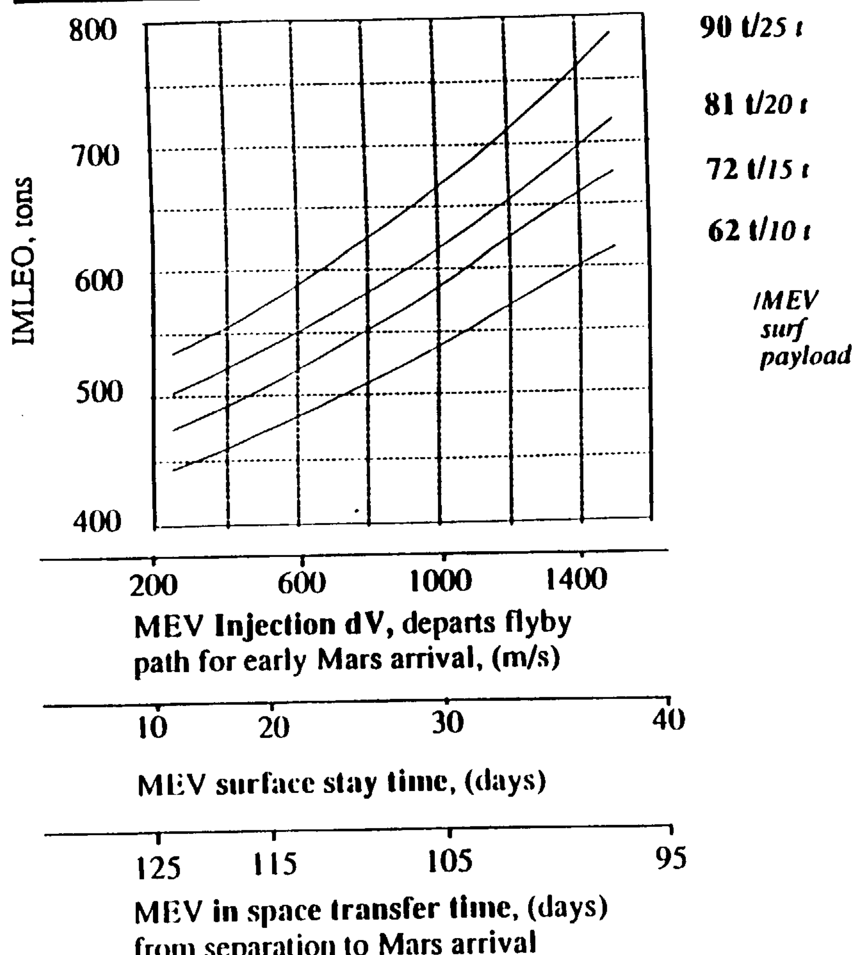
IMLEO vs MEV Wt & Injection dV for Early Arrival

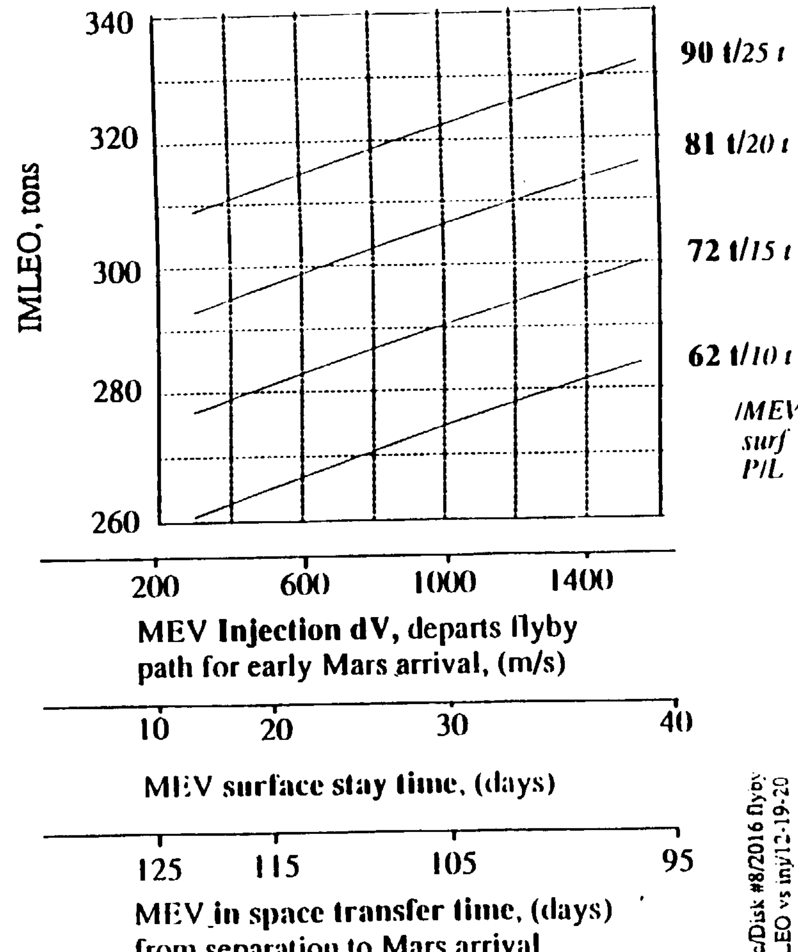
***Option 2**: Utility stage rendezvous w MTV on hyperbolic swingby path*

ADVANCED CIVIL SPACE SYSTEMS

BOEING

year 2016,
499 day trip time





* *Asc veh ascends to rendezvous w MTV on hyperbolic swingby path, thus MEV wt is function of swingby Vhp*

Mac/Disk #8/2016 flyby
IMLEO vs inj/12-19-20

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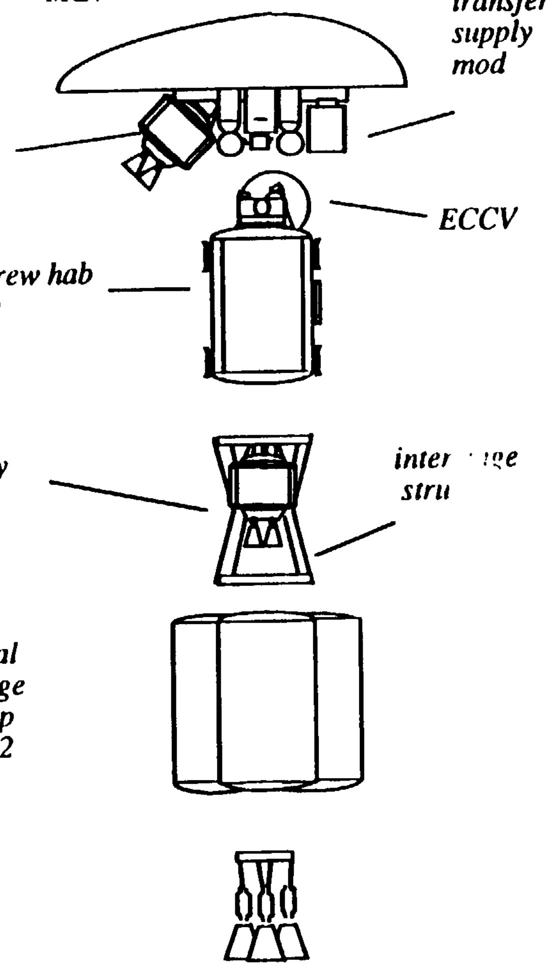
ADVANCED CIVIL SPACE SYSTEMS

Mars Flyby with Surf Mission

Option 2: *ECCV ret, Crew of 4, TMI dV= 4271 m/s, Powered swingby dV = 806 m/s*

BOEING

year 2016
499 day trip



preliminary configuration; not necessarily to scale

	Element	mass (kg)
[378]	MTV crew hab module 'dry'	27567
[398+371]	MTV hab consumables & resupply	5156
sum	***MTV crew hab total***	**32723**
[230]	***ECCV***	**7000**
[128]	Powered swingby stg usable propellant	8362
[150+151]	Powered swingby stg outbound boiloff	876
sum	*Total TEI propellant*	9238
[118]	RCS propellant	454
[121]	Outb midcourse correction prop	580
[122]	Inb midcourse correction prop	473
[161]	Powered swingby stg inert	3871
sum	***MTV propulsion stg total***	**14616**
[78]	MEV Mars capture & desc aerobrake	17508
[65]	MEV asc vehicle	23629
	MEV descent stage	23863
[77]	MEV surface cargo	25000
[106]	***MEV total***	**90000**
	MEV transfer leg supply module	4000
[183+203]	MEV kick stg/Mars orbit dep stg inerts	8545
[180]	MEV kick stg propellant	13646
(653)	Mars orbit dep propellant	20669
[232]	***MEV kick stg total***	**42854**
[i364]	MTV-TMI interstage wt	500
[172-173]	TMI inert stage wt	38110
[173]	TMI propellant load	342990
[172]	***TMI stage total***	**381100**
[171]	***IMLEO***	**572793**

Veh synthesis model run# marsflyby.dat;23
Mac:/disk #13/ flyby mission chem sum wt statement/1-2-91
/STCAEM/bd/11Jan91

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Mars Flyby/'Dash' mission

2020 IMLEO vs MEV Wt & Injection dV for Early Mars Arrival

Vehicle IMLEOs were calculated for the Mars Flyby/Dash mode to determine the savings in vehicle weight over the traditional stopover opposition missions. Four vehicle propulsion types were analyzed:

MEV	TMI stage	powered swingby stage	column
1.	Chemical O_2/H_2	Chemical O_2/H_2	left
2.	NEVA NTR	Chemical O_2/H_2	right

MEV weight was varied to show IMLEO delta's to the MEV cargo delivered (to Mars surface) weight. For this chart Ascent departure from Mars orbit to rendezvous *Option 1* was utilized.

O2/H2 chem & NERVA NTR+O2/H2 chem:: Mars Flyby with Surf Exploration:

IMLEO vs MEV Wt & Injection dV for Early Arrival

Option 1: Ascent vehicle direct rendezvous w MTV on hyperbolic swingby path

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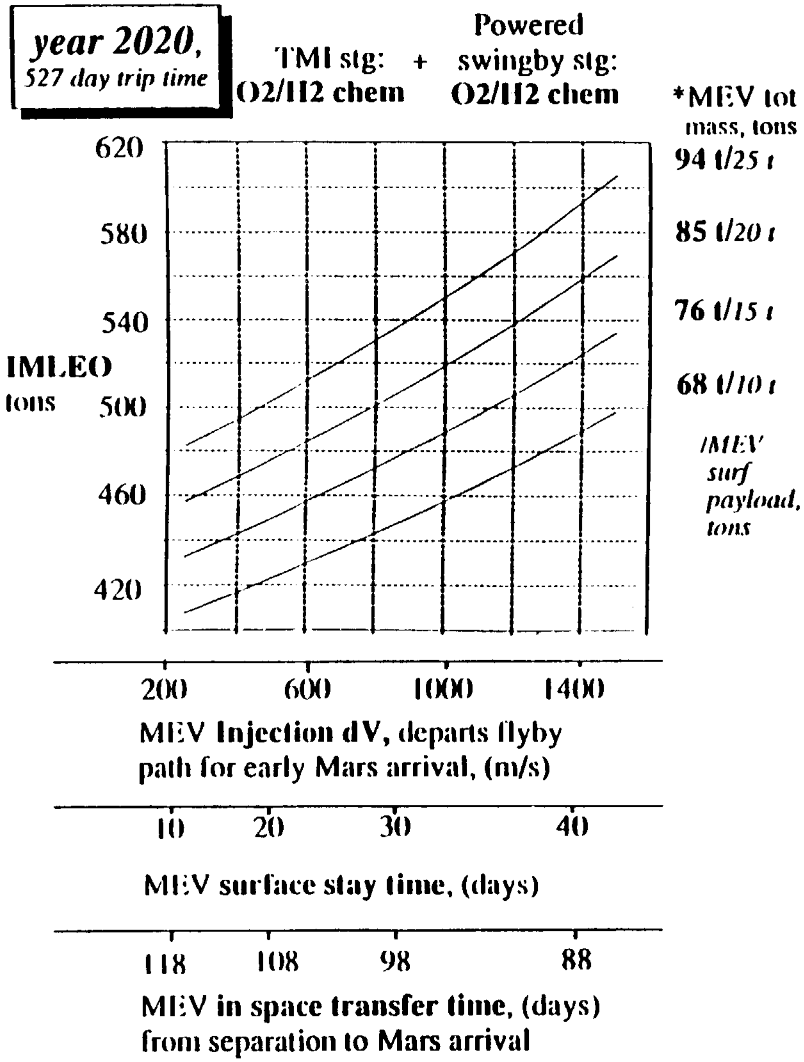
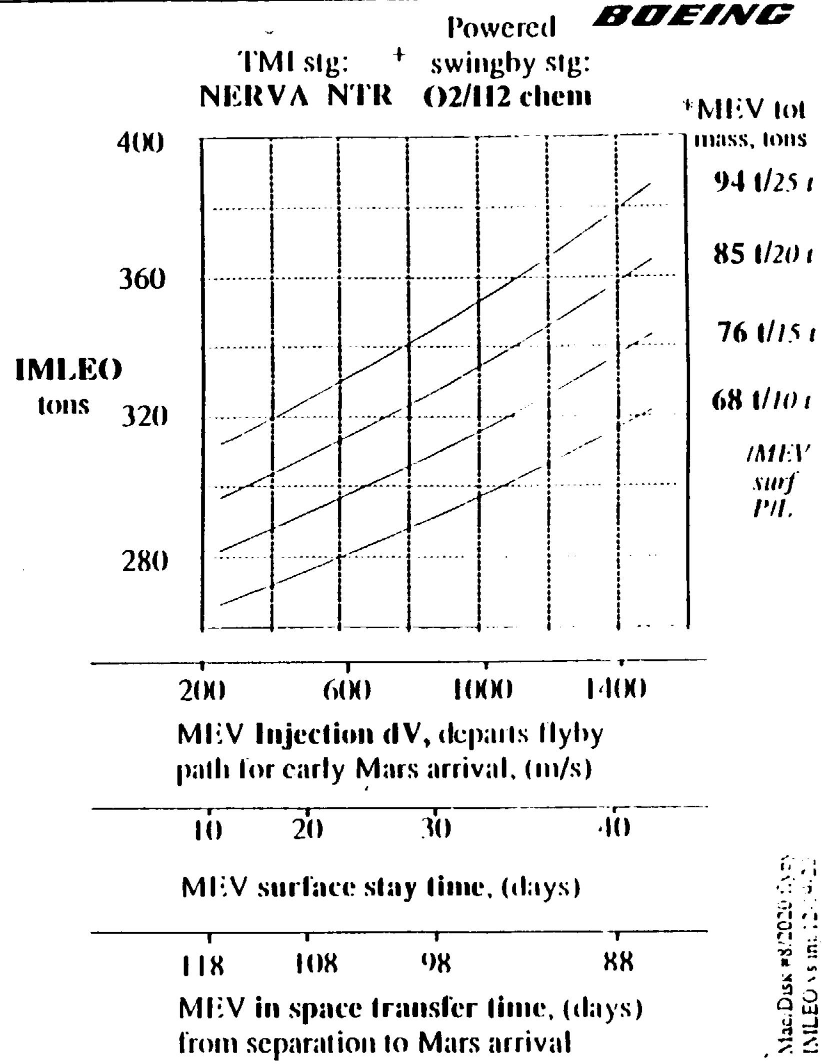
year 2020,
527 day trip time





Mac.Disk #8/2020 ...
IMLEO vs ...

* *Asc veh ascends to rendezvous w MTV on hyperbolic swingby path, thus MEV wt is function of swingby Vhp; = 3917 (m/s) for 2020*

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Mars Flyby/'Dash' mission

2020 IMLEO vs MEV Wt & Injection dV for Early Arrival

Vehicle IMLEOs were calculated for the Mars Flyby/Dash mode to determine the savings in vehicle weight over the traditional stopover opposition missions. Four vehicle propulsion types were analyzed:

	TMI stage	powered swingby stage	column
1.	NERVA NTR	NERVA NTR	left
2.	Advanced NTR	Advanced NTR	right

MEV weight was varied to show IMLEO delta's to the MEV cargo delivered (to Mars surface) weight. For this chart ascent vehicle Mars departure *Option 1* was utilized.

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NERVA NTR and Advanced NTR: Mars Flyby Missions with Surface Exploration:

IMLEO vs MEV Wt & Injection dV for Early Arrival

Option 1: Ascent vehicle direct rendezvous w MTV on hyperbolic swingby path

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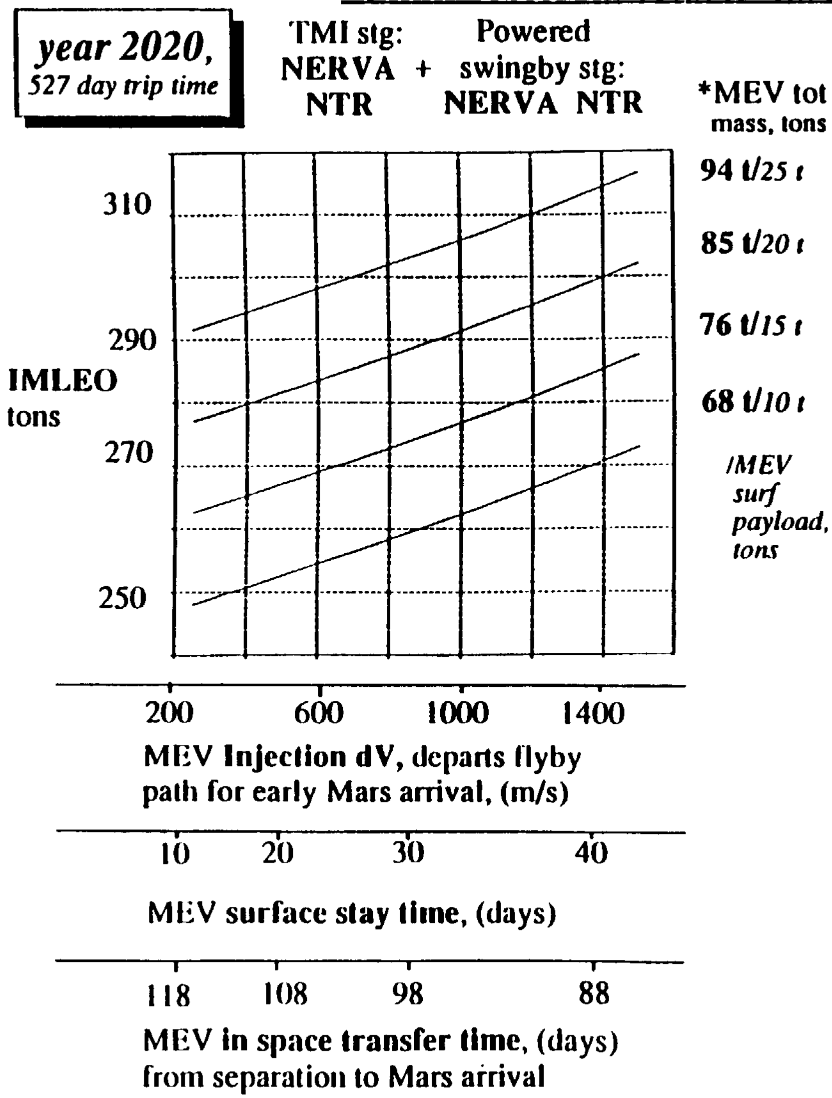
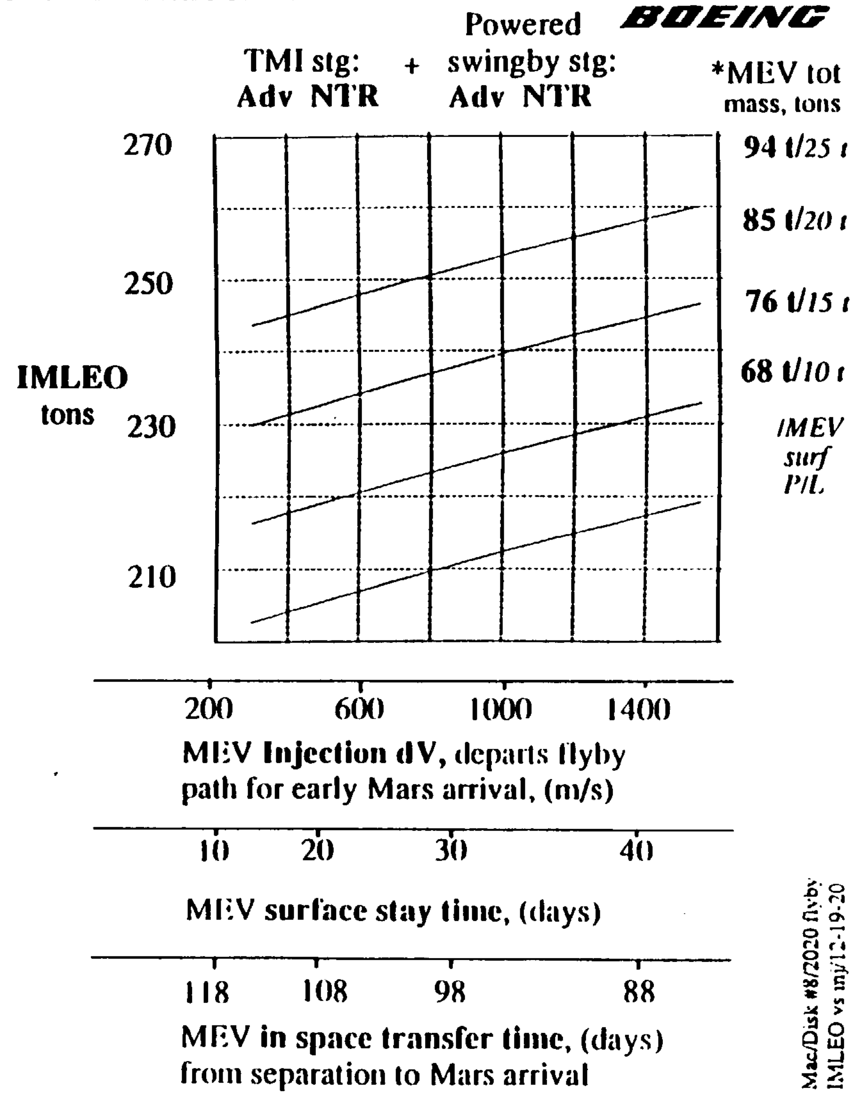
year 2020,
527 day trip time





Mac/Disk #8/2020 flyby
IMLEO vs inj/12-19-20

* *Asc veh ascends to rendezvous w MTV on hyperbolic swingby path, thus MEV wt is function of swingby Vhp; = 3947 (m/s) for 2020*

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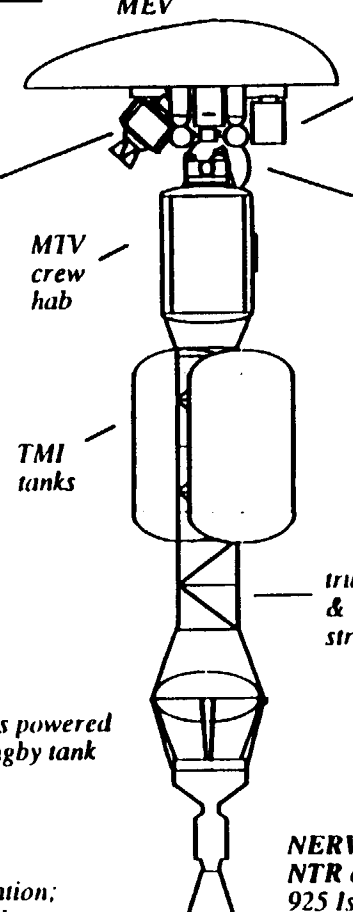
Mars Flyby with Surf Mission: NERVA NTR TMI & Powered Swingby Stage

Option 1, ECCV ret, TMI dV= 3871 m/s, Powered swingby dV =1662 m/s, kick dV =500'

2020 *527 day trip*
20 day surf stay



preliminary configuration; not necessarily to scale

Element	*Chem kick stg*	*NTR main stg*
	MEV injection for early arrival done by	
MTV crew hab module 'dry'	27648	27648
MTV hab consumables & resupply	5843	5843
MTV crew habitat module total	**33491**	**33491**
ECCV	**7000**	**7000**
Powered swingby stg usable propellant	21085	9926
Powered swingby stg outbound boiloff	1271	2210
Total Powered swingby propellant	23356	12136
RCS propellant	504	504
Outb midcourse correction prop	775	775
Inb midcourse correction prop	525	528
Powered swingby propul stg inert	7961	5563
Powered swingby propulsion stg total	**33121**	**19506**
MEV Mars capture & desc aerobrake	12302	12302
MEV asc veh 1st stg (drop tanks & prop)	11258	11258
MEV asc veh with 2nd stg prop	15962	15962
MEV descent stage	29836	29836
MEV surface cargo	25000	25000
MEV total	**94358**	**94358**
MEV transfer leg supply mod	4000	4000
MEV kick stg (early arrival) inerts	4094	0
MEV kick stg propellant	12457	0
MEV kick stg total	**16542**	**0**
NTR main stage slowdown burn prop		6156
NTR engine mass (engT/W=3.5)	9684	9684
NTR engine radiation shield mass	2500	2500
NTR stg tank struts mass	1000	1000
TMI inert H2 tank wt (14% t fraction)	19007	16790
TMI propellant load (H2)	116758	102830
TMI stage total	**135766**	**119620**
IMLEO	**337462**	**297315**

Veh synthesis model run# marsflyby.dat;29,31
Mac:/disk #13/2016 flyby mission chem sum wt statement/12-22-90
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2016 and 2020 Flyby/dash vs Stopover Mission IMLEO Comparison

Vehicle IMLEOs were calculated for the Mars Flyby/Dash mode to determine the savings in vehicle weight over the traditional stopover opposition missions. Four vehicle propulsion types were analyzed:

	TMI stage	powered swingby stage	mode	MEV cargo
1.	Chemical O2/H2	Chemical O2/H2	expendable	25 tons
2.	NEVA NTR	Chemical O2/H2	expendable	25 tons
3.	NERVA NTR	Chemical O2/H2	expendable	25 tons
4.	Advanced NTR	Advanced NTR	expendable	25 tons

Results: **2016** Flyby/dash savings over Stopover missions:

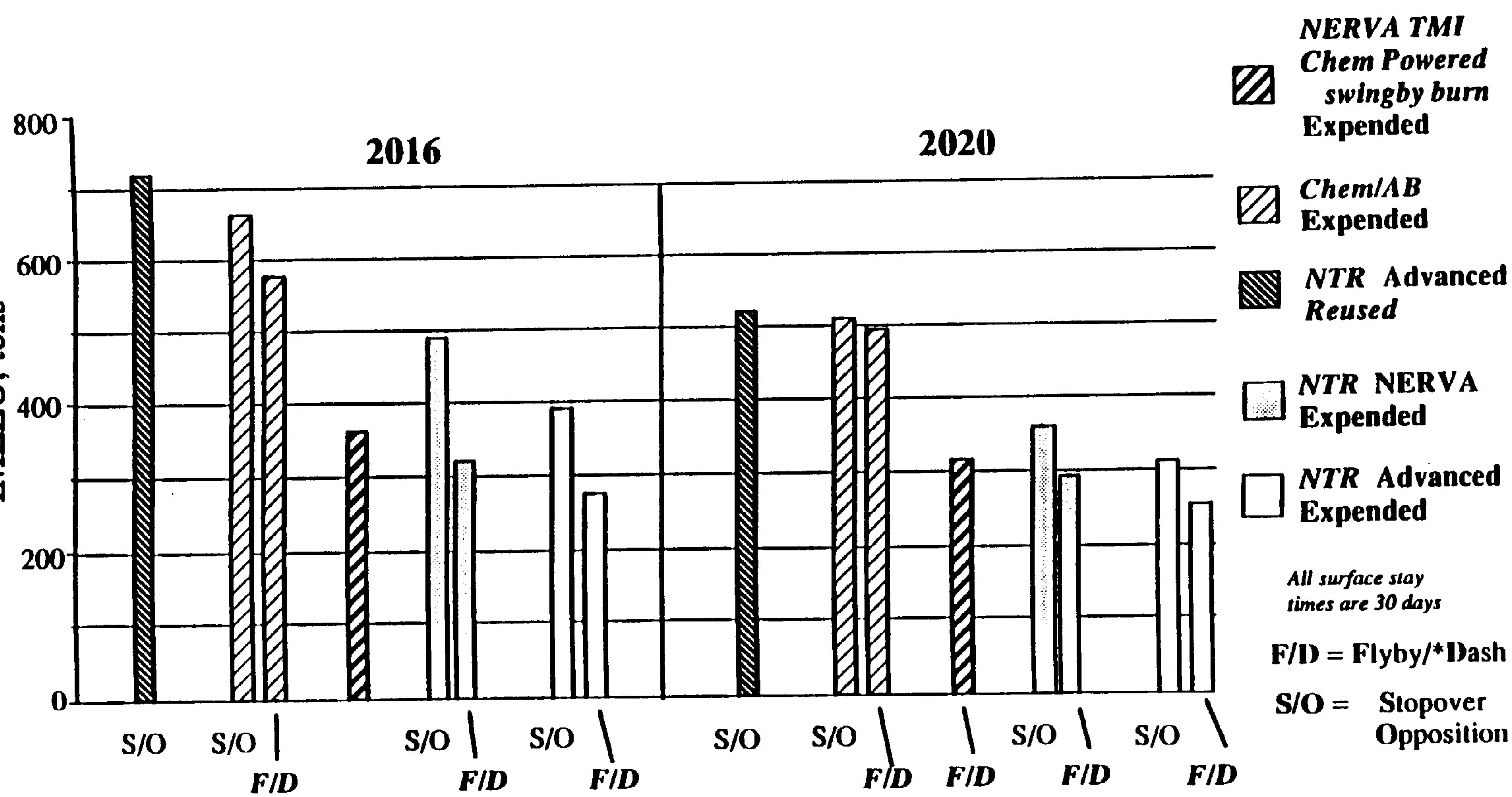
- The all chemical O2/H2 vehicle: saves 75 tons IMLEO (12%)
- The all NERVA NTR vehicle: saves 165 tons IMLEO (34%)
- The Advanced NTR vehicle: saves 115 tons IMLEO (29%)

Results: **2020** Flyby/dash savings over Stopover missions:

- The all chemical O2/H2 vehicle: saves 10 tons IMLEO (even)
- The all NERVA NTR vehicle: saves 70 tons IMLEO (19%)
- The Advanced NTR vehicle: saves 60 tons IMLEO (19%)

Flyby/Dash vs Stopover Oppositions Missions

with Venus Swingbys. MEV delivers 25 tons to the surface



*Opposition Flyby mission with MEV 'Dash' (small deep space burn velocity addition) to arrive at Mars early for surface exploration before MTV stg arrivial & rendezvous

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2016 Flyby/'Dash' Opposition Mission

MEV delivers 25 tons to surface

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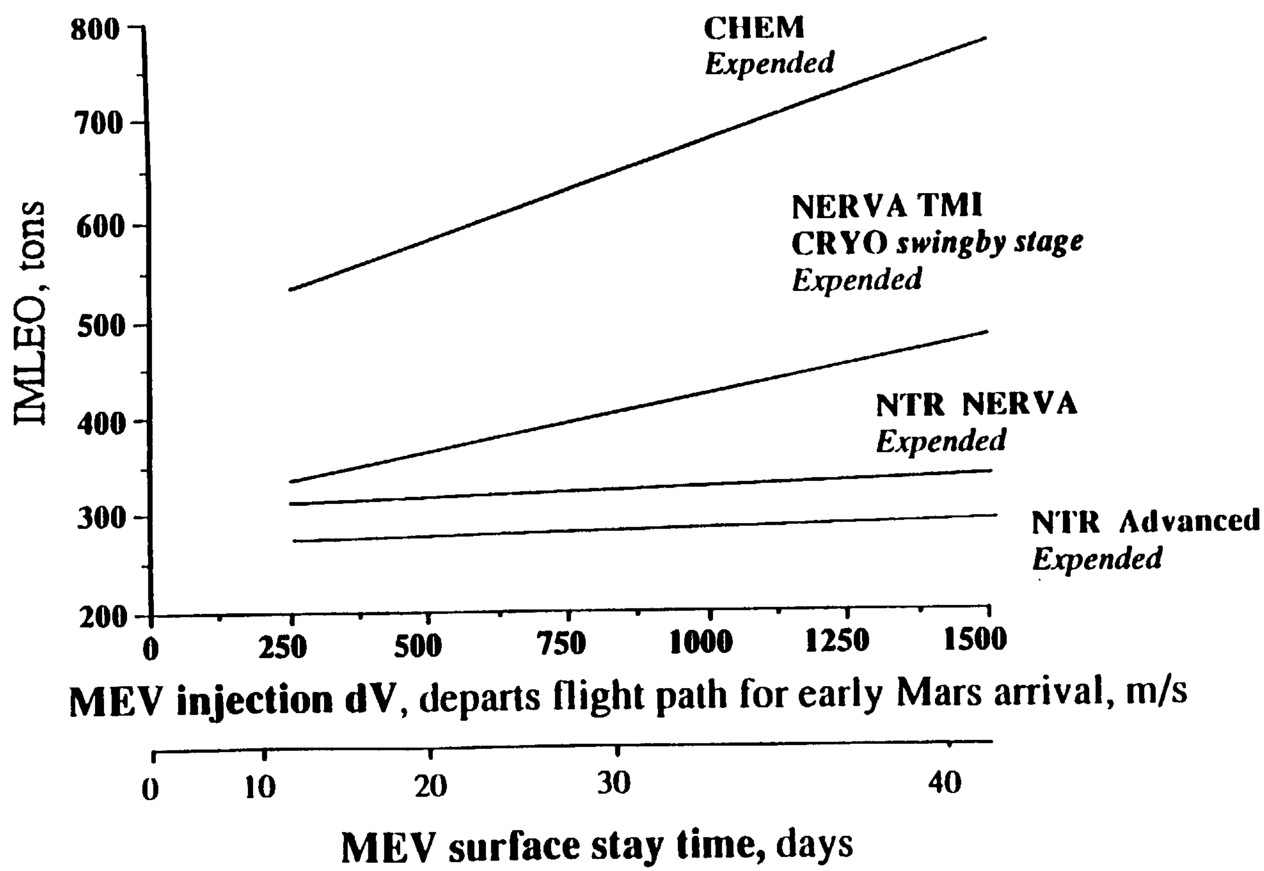

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Mac/Disk #5/IMLEO/yr bar chart/1 10 91

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Architecture Matrix

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Reference Matrix to Alternative Architectures

In considering a complex task, it is useful to organize it into a heirarchy of levels. The higher levels are more important or more encompassings, while the lower levels include more detail or are more specific. Constraints (e.g., requirements and schedules) flow down from the higher levels and solutions or implementations build up from the lower levels. The first figure shows a heirarchy of six levels from national goals to performing subsystems. The following section discusses the fourth level, exploration architectures, in terms of the lower levels: element concepts and performing subsystems. Selection of preferred architectures will require the Government (the National Space Council, the President, and the Congress) to first define the top three levels.

Implementation Architectures

Seven architectures have been selected for examination: four different propulsion types (Cryogenic/Aerobrake, NEP, SEP, and NTR); two variations of In-Situ Resource Utilization (ISRU) for propellants with Cryogenic/Aerobrake propulsion (Lagrange point 2 refueling and Mars surface refueling); and a cycling spacecraft concept. Three basic levels of program scope are identified: small, moderate, and ambitious.

Multiple options can be generated within the basic architectures, varying launch vehicle capacity, orbital node type, and mission profile and propulsion type for the various Lunar and Mars vehicles.

Aerobraking is found to be applicable to all seven architectures, placing it as a 'critical' technology. Electric propulsion leads to the lowest reference vehicle mass, and also almost the lowest resupply mass. ISRU/Cryo leads to the lowest estimated resupply mass since most of the propellant is derived locally rather than coming from Earth.

Cost Models

Cost estimation is being performed using "parametric" methods. This technique uses a parameter, usually weight, as an input to empirically derived equations that relate the parameter to cost. It should be recognized that the source data for the cost models is past program experience, while the hardware being estimated will be built one or two decades from now. Therefore these cost estimates should be assumed to have a standard deviation on the order of +-100%. Hardware at technology readiness level 5 may be assumed to have a standard deviation in cost estimate of +-30%. No revenues from sale of products, services, or rights (i.e. patent rights, data rights), or commercial investment, are assumed in the cost estimates. These might appear in a scenario such as the Energy Enterprise.

Aa an example, the cost estimate for a NEP architecture shows an average annual funding level of $8 billion per year after initial ramp-up.

The principal cost drivers identified include number of development projects, reuseability, mass in Earth orbit, and mission/operational flexibility.

Analysis Methods

Individual trade studies are performed within each architecture to optimize it against evaluation criteria. The principal evaluation criteria to date has been initial mass in low Earth orbit, as a proxy for cost. The results of this optimization will then be compared to each other in groups. The early Mars group will compare all-propulsive, aerobraking,

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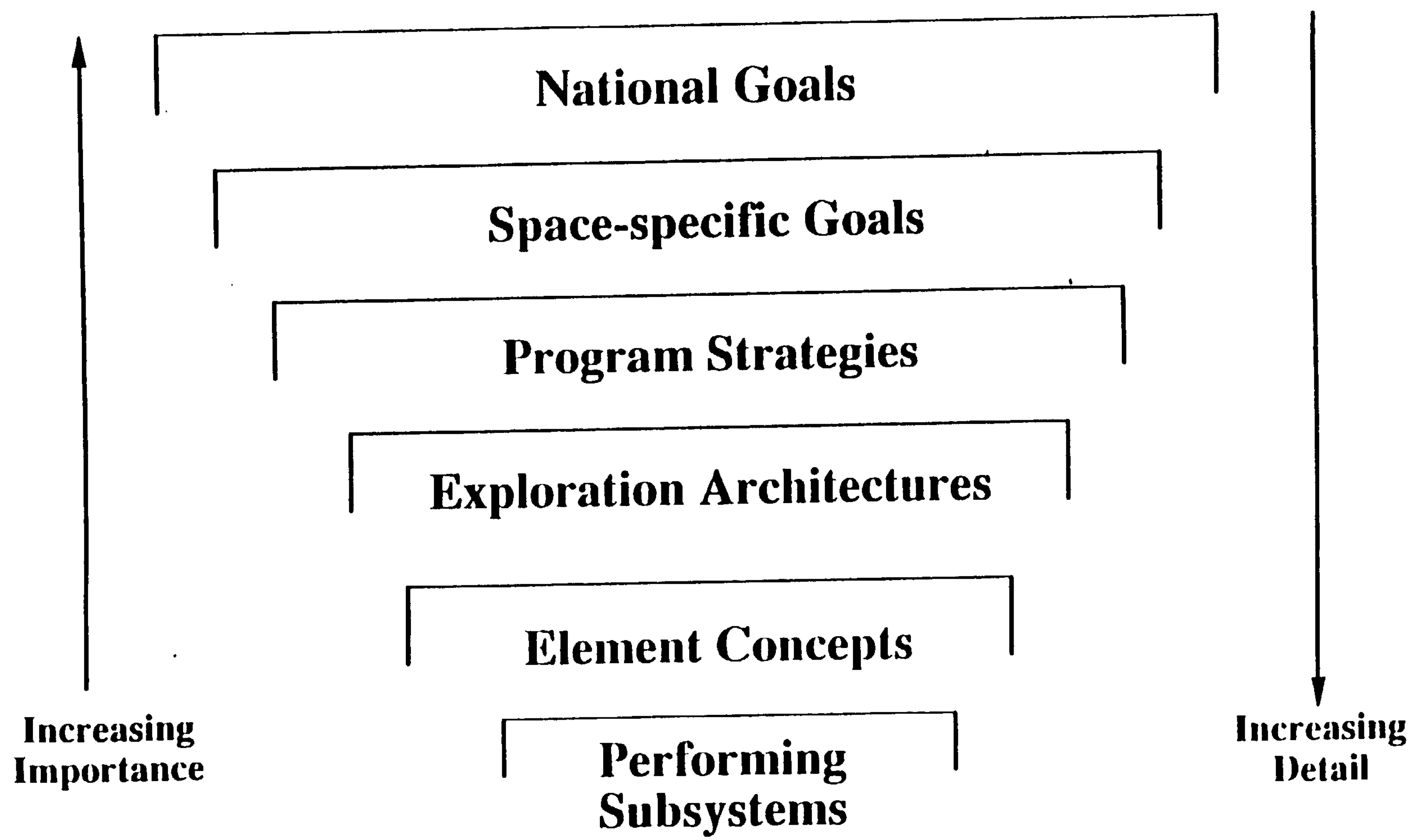
direct travel, and nuclear thermal among themselves. The electric propulsion group will compare SEP and NEP. The innovative group will compare Lunar oxygen to cycler orbits. These concepts may both be retained if it is advantageous to do so. Finally, the choice between early Mars and Late/Evolving Mars will need to be made on the basis of cost, risk, and performance, while combining the best features from each group.

Logical Types for Space Programs

Architectural planning for a space program deals with many levels of information. A major space program like the space exploration initiative must respond directivity to national goals in traceable ways. While we do not determine national goals, it is our business to understand how exploration architectures can be evaluated in terms of national goals.

National goals translate to space specific goals for specific exploration programs such as science emphasis or expanding human presence. These in turn can lead to program strategies for space-specific goals such as low risk, high technology, low cost and so forth. Finally, exploration architectures are integrated assemblages of systems, mission profiles, and operations, necessary to satisfy program goals.

"Logical Types" for a Space Program



Each logical type subsumes all the subordinate types

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Overall Study Flow

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown on the facing page. Combinations of major technologies, such as electric propulsion and aerocapture, were quickly determined to be uneconomic in view of high development costs. Further, we found that electric propulsion systems could perform both crew and cargo Mars missions if crews are transported to and from the electric system at about lunar distance by a lunar transfer vehicle.

New systems introduced included nuclear thermal rocket (NTR) and Mars direct. NTR was introduced as an option by NASA during the "90-day study". We introduced the Mars direct profile (everything is landed on Mars; the return propulsion system is loaded with oxygen and perhaps fuel as well on Mars) in March 1989. Martin-Marietta subsequently publicized one variant of this concept.

Lunar oxygen for Mars missions was found to be uneconomic because of long payback time for the launch mass required to emplace lunar oxygen production on the Moon. Lunar oxygen has a reasonable return on investment for lunar transportation at two or more lunar trips per year.

The cycler architecture was broadened to include semi-cyclers. Late in the study we introduced an NTR-dash mode (described later in this briefing) closely related to the semi-cyclers.

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Overall Study Flow

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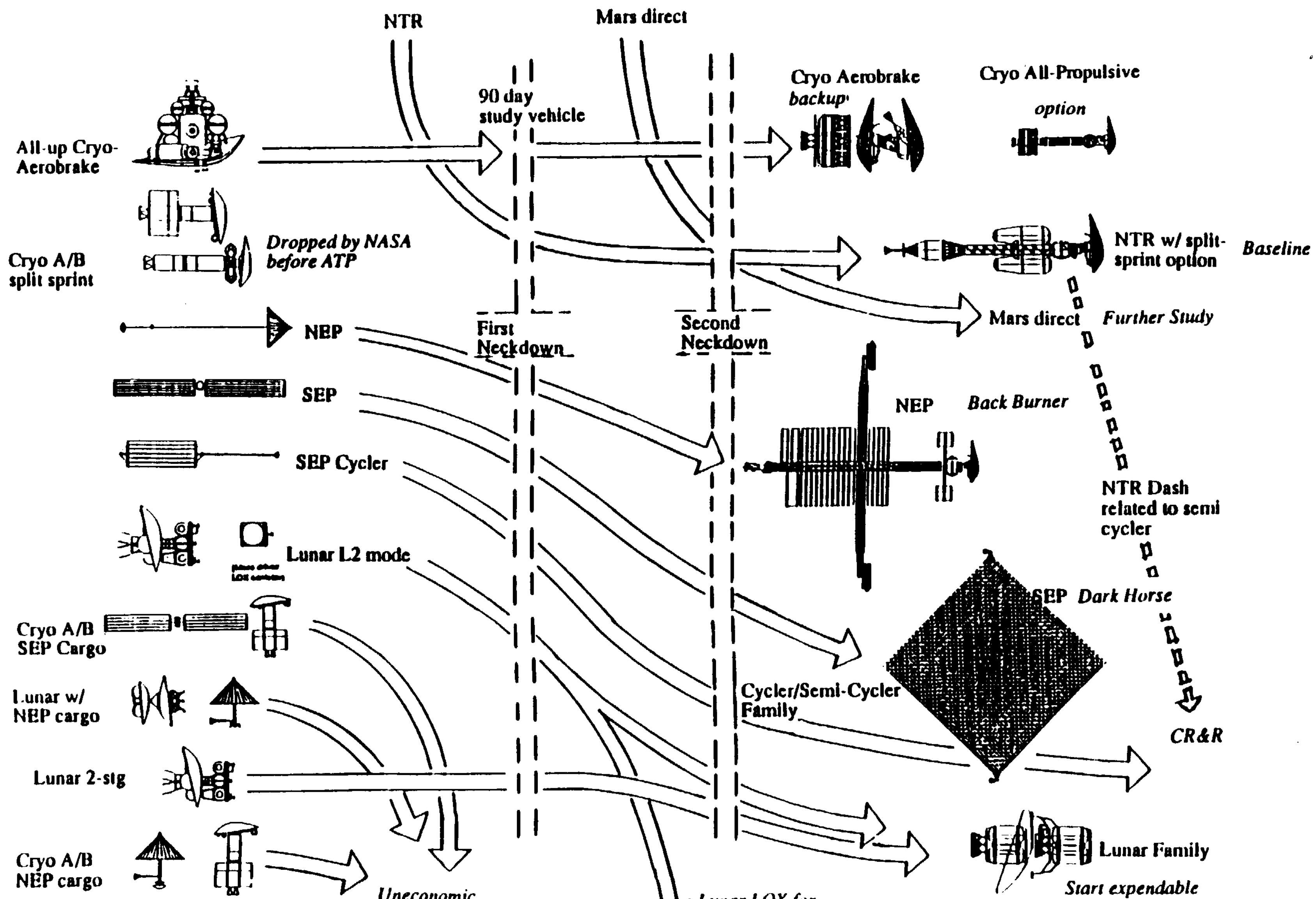

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Program Implementation Architectures

We have selected seven program implementation architectures for architectural analysis. These seven architectures incorporate the advanced propulsion options of principal interest in complete evolutionary architectural scenarios for lunar and Mars exploration. The facing page lists the features of each architecture and the rationale for selection of each.

Some of the architectures include suboptions. For example, the nuclear electric propulsion and solar electric propulsion architectures include optional use of the electric propulsion system for lunar cargo delivery from LEO to lunar orbit. The L2-based cryogenic aerobraking architecture includes use of NTR and NEP vehicles for LEO to L2 cargo delivery as options, and also includes a cryogenic all-propulsive conjunction mission option.

Program Implementation Architectures

Architecture	Features	Rationale
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	NASA 90-day study baseline
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.	High performance of nuclear electric propulsion
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.	High efficiency of solar electric propulsion; find cost crossover for array costs.
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.	High Isp of nuclear rocket enables avoidance of high-energy aerocapture at Mars.
L2 Based cryogenic/ aerobraking	L2-based operations; use of lunar oxygen.	L2 base gets out of LEO debris environment. Lunar oxygen reduces resupply by ~ factor 2.
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	Eliminates Mars orbit operations.
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	Eliminates boosting massive Mars transfer vehicle.

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SEI Program Scopes for Transportation Architecture Analysis

There are many space-specific goals and program strategies. We believe that transportation architectures will respond mainly to program scope. Some architectures are best suited to small program with early goals and others best suited to long range larger programs with ambitious goals. We have selected three representative scopes for small, moderate and large programs as illustrated on the facing page. These scopes permit definition of transportation requirements in terms of numbers of people and amounts of cargo transported to particular locations on particular schedules.

The second important feature of the scopes we intend to investigate is that they cover a scale factor greater than ten. A man tended science station may have few people on the Moon for short periods, or few people on Mars for short periods every other year. Permanent science bases will involve a dozen or so people. Industrial development of lunar resources on a scale of helium-3 scenarios leads to numbers of people presently estimated in the range of thousands by 2050. Beginnings of humans settlement of Mars involves numbers in the range hundreds to thousands. The 20-25 horizon for SEI is expected to permit growth in numbers of people only to dozens or so.

SEI Program Scopes for Transportation Architecture Analysis

Descriptor	Small	Moderate	Ambitious
Lunar Operations	Man-tended science station	Permanent science base 6 - 12 people	Industrial development of lunar resources
Mars Operations	Expeditionary visits ~4 people	Permanent science base 6 - 12 people	Beginnings of human settlement

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Three Activity Levels for Architecture Evaluation

We established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in-space transportation technologies as baselines for greater activity levels.

Activity levels were selected with underlying program objectives in mind:

(1) The minimum lunar program establishes astrophysical observatories on the Moon and provides a man-tending capability to maintain them. To the extent that man-tending lunar visits are not needed for the observatory system, the transportation capability can be used to explore interesting lunar sites for lunar geoscience objectives.

(2) The minimum Mars program is very similar to Apollo, i.e. six sites visited for short periods (two sites per mission and three missions); samples obtained within a few km. of each landing site. If the manned visits are preceded by suitable robotic missions, the scientific payoff for these visits can be high relative to the investment.

(3) The "full science" lunar program adds human permanence at the Moon for extensive scientific and technological exploration. Where the minimum program offers very little opportunity for lunar geoscience, this program offer much. It also permits development of in-situ resource technology for production of surface systems. The reference program also emplaced a lunar oxygen production system to serve the transportation system.

(4) The "full science" Mars program multiplies by several the crew person-days on Mars by including more missions and by more staytime per mission. This program falls short of a permanently-occupied base on Mars, but achieves surface stays greater than a year.

(5) The lunar industrialization program adopts production of helium-3 as a strawman industrial objective and places enough facilities and infrastructure on the Moon by 2025 to return 1 GWe helium-3 fusion fuel to Earth.

(6) The Mars settlement program moves towards Mars settlement. A robust nuclear electric propulsion system is fielded, with convoy flights by 2015. Mars population reaches 24 by 2025, and the transportation system is capable of increasing Mars population by 24 per opportunity by 2025.

Three Activity Levels for Architecture Evaluation

Minimum	Median (full science)	Industrialization /settlement
Just enough to meet President's objectives	*Meet science objectives of lunar/Mars exploration*	*Return of practical benefits to Earth*
• Permanent lunar facilities, not permanent human presence	• Human permanence	• Extensive facilities and infrastructure on the Moon by 2025
• Astrophysics observatories • Man-tending capability • Explore interesting sites	• Opportunity for lunar geoscience • In-situ resource technology	• Lunar population 30 by 2025
• Three missions to Mars	• Order of magnitude more crew time on Mars	• Mars population 24 by 2025
• Similar to Apollo • Two sites per mission • Samples within a few km. of landing sites	• Approaches permanent base (stay time >1 year)	• Capable of increasing Mars population by 24 per opportunity by 2025.

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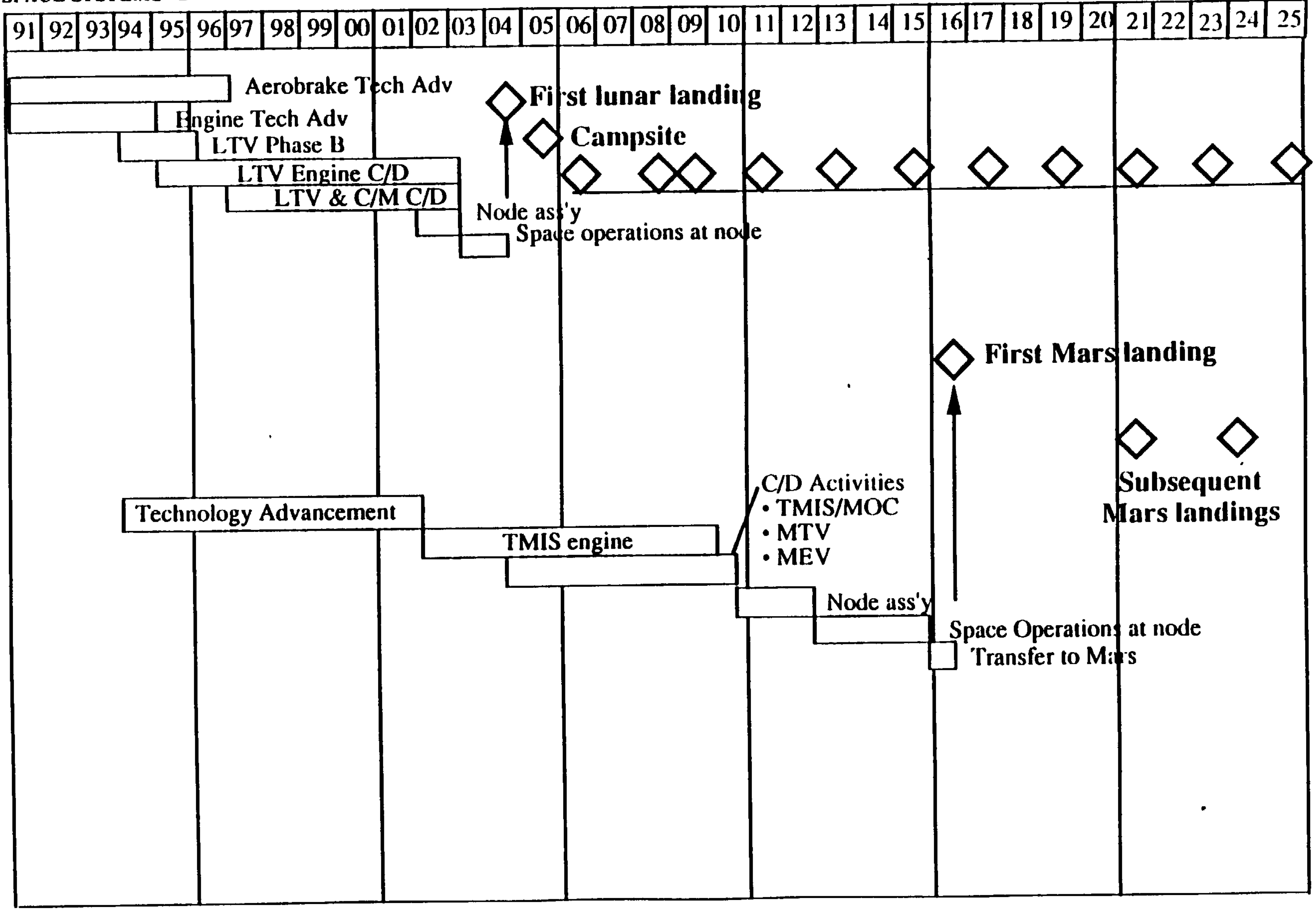
Minimum Program

The minimum program reference averages about 1/2 lunar trip per year and has only three Mars missions. Lunar science facilities are man-tended. Each Mars mission carries two landers (MEVs) for added exploration capability and a measure of rescue capability. Surface stays are about 30 days. Lunar and Mars in-space transportation systems are expendable.

Minimum Program



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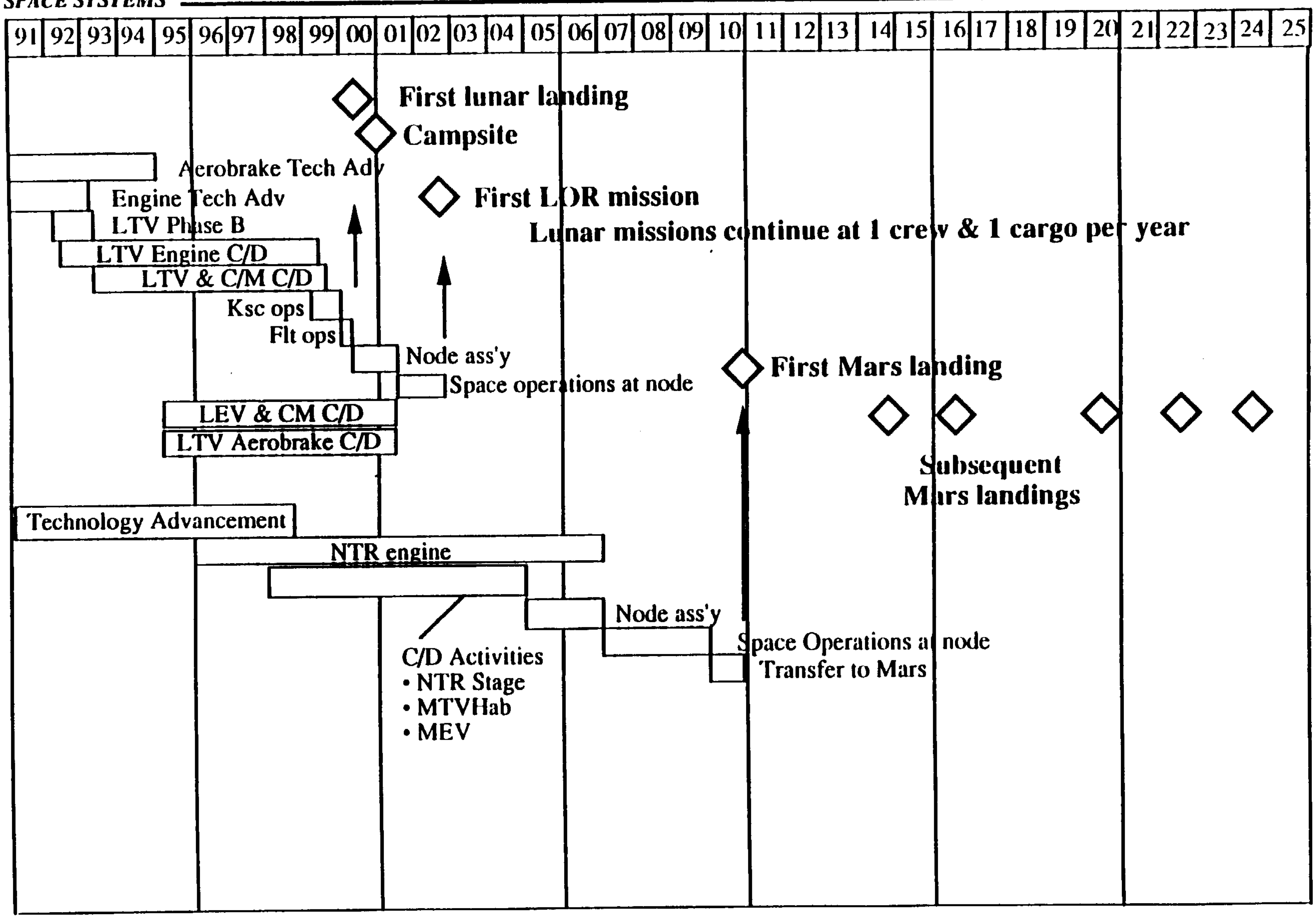
Full Science Program

The full science program reference has about 2 lunar missions per year, to establish permanent human presence on the Moon with adequate supplies and equipment for extensive science and exploration. Lunar oxygen for lunar transportation is introduced about mid-way through the lunar program. Six Mars missions are accomplished, with later missions staying on Mars for more than a year. The Mars missions use multiple landers, as many as four late in the program.

Full Science Program



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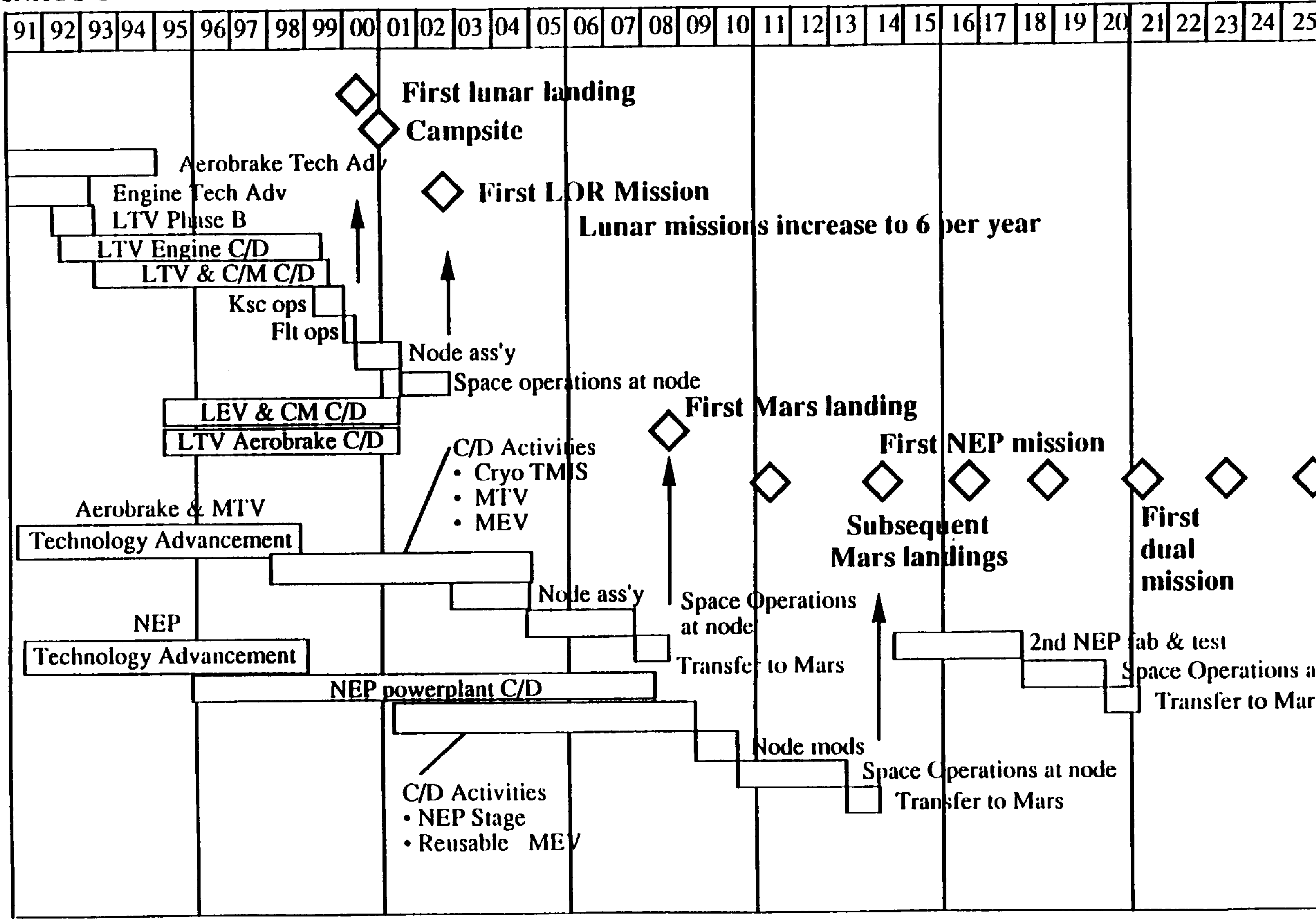
Industrialization and Settlement Program

The industrialization and settlement program is very aggressive for both the Moon and Mars. Thousands of tons of industrial equipment are delivered to the Moon, driving lunar cargo trips up to five per year. Lunar oxygen is placed in production as early as possible. One crew trip per year leads to a population of 30 because crew stay times on the Moon increase to several years.

Initial Mars missions use a cryogenic/all-propulsive system because the aggressive nature of the scenario merited an initial Mars mission as early as possible, and the reference nuclear electric propulsion system cannot be ready in time. The NEP missions are operated in a crew rotation/resupply mode, opposition profile, with each crew staying one synodic period (about 2.2 years). The reference scenario evolves to reusable MEVs based on Mars, fueled from Mars resources. Heavy cargo capability is provided, up to 250 t. per opportunity by 2020. The Mars population grows to 24, and by the end of the scenario can continue to grow by 24 or more per opportunity.

Industrialization and Settlement Program



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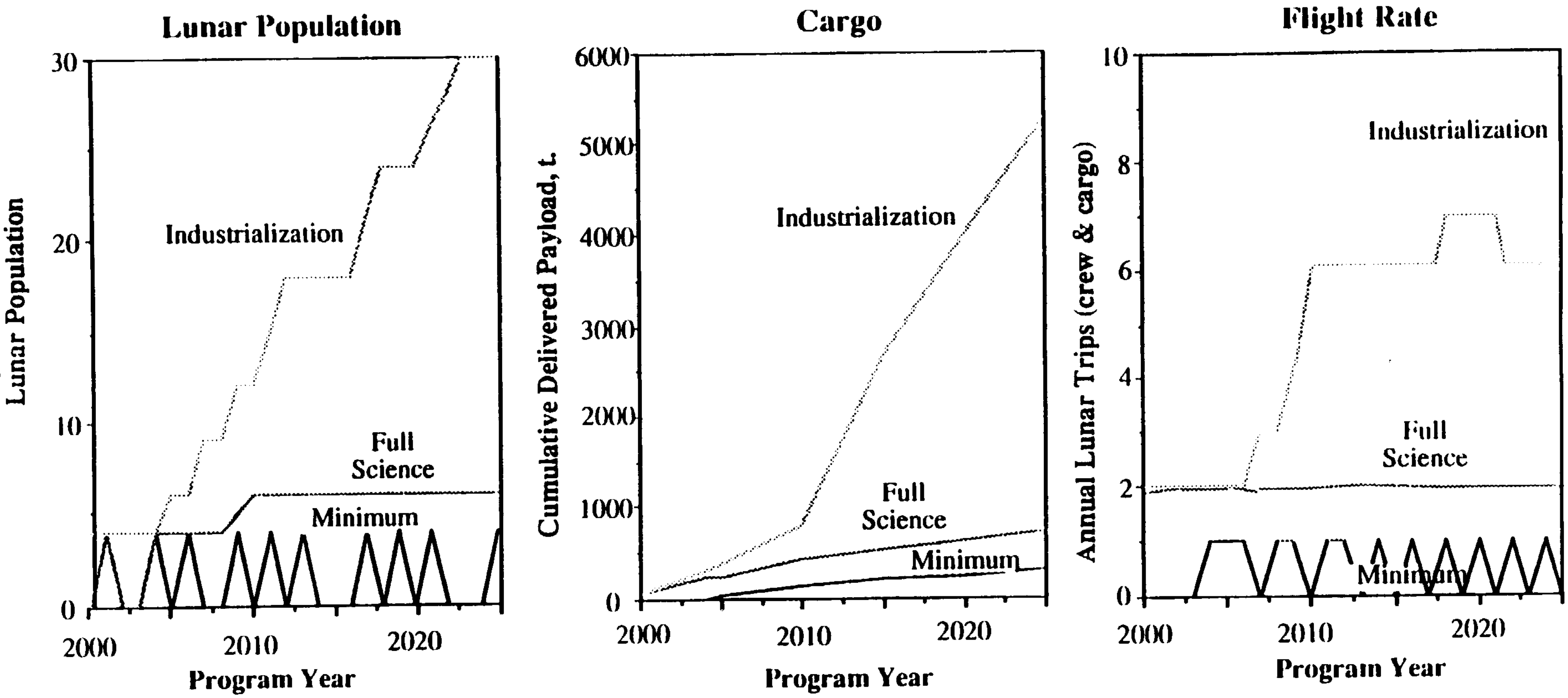
Lunar/Mars Program Comparisons

The next two charts compare the lunar and Mars program scenarios in terms of population, cumulative cargo delivered, and flight rate. The lunar population for the minimum scenario is four people for 30 to 40 days about every other year. The Mars population for the minimum scenario is 6 people on each of 3 conjunction missions, with 30 to 40 day surface stays. The full science menu scenario grows to year-long surface stays on conjunction missions. The lunar industrialization program goes to long stay times with indigenous food growth to build population. The Mars proto-settlement program obtains continuous presence by operating the NEP on an opposition-like profile in crew rotation/resupply mode. Later in this scenario, a second NEP is operated to provide two trips to Mars each opportunity.

These scenarios were the "input" to the manifesting and life cycle cost analyses.

Lunar Program Comparison



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Mars Program Comparisons

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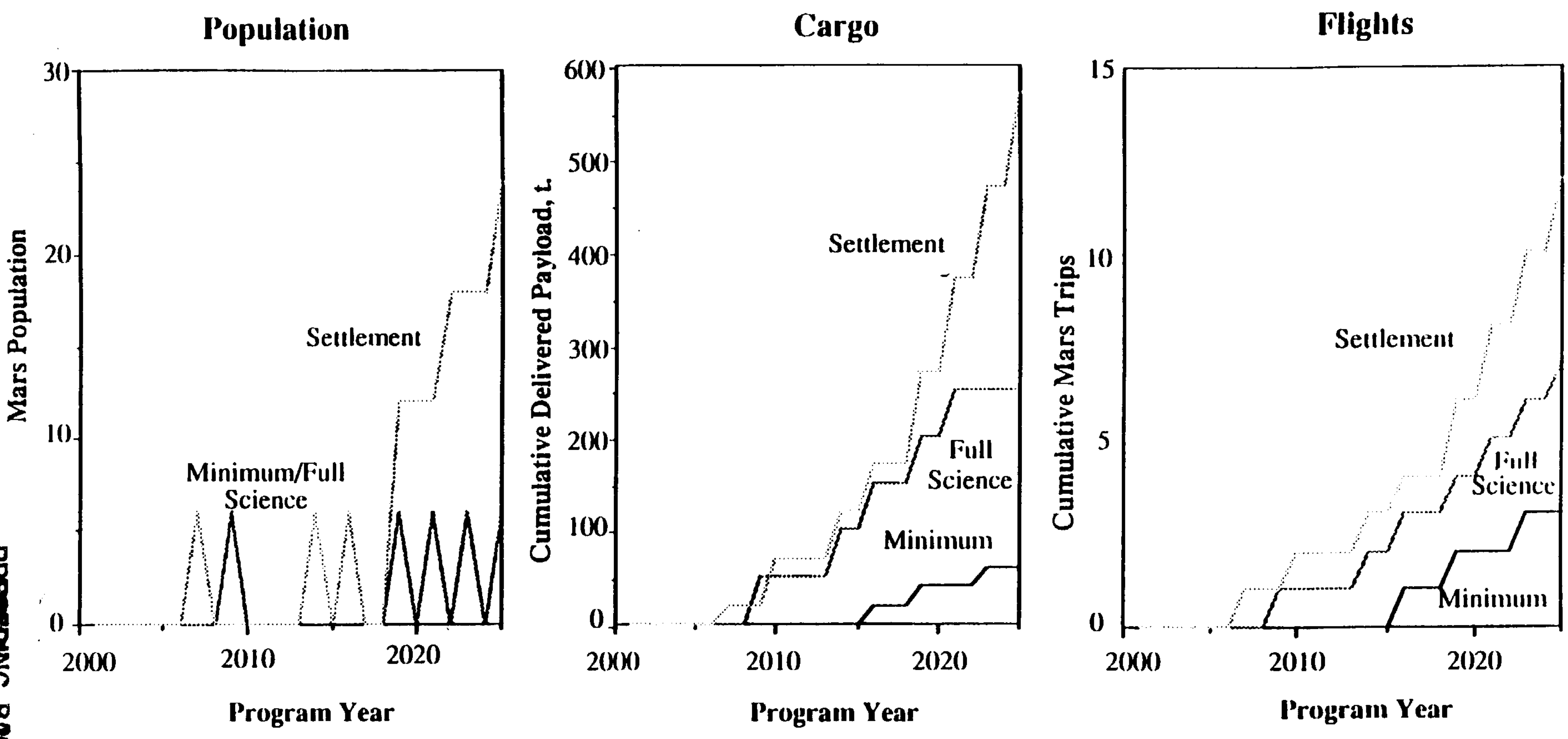

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Architecture/Launch Vehicle/Node Trends

Issues

- Launch vehicle size, shroud size, and lift capacity.
- Node complexity and cost.
- On-orbit assembly complexity
- Number of launches per year
- Development cost
- Per-mission cost

Trends from Architecture Analyses

- Large launch vehicle (up to 300 t. lift) does not eliminate on-orbit assembly.
- Keys to on-orbit assembly are (1) design the vehicle to keep it simple; (2) design for automation and robotics; (3) reusable space vehicles to reduce the frequency of assembly operations.
- Advanced in-space transportation technology reduces launch requirements enough that a 100-t., 10-meter shroud launch vehicle is adequate.
- Ultra-large launch vehicle results in high early program costs and is much more costly than advanced in-space transportation technology.
- Evolution and design for evolutionary transitions are the keys to affordable, efficient programs with long-term growth.

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Available Options

The facing page is a typical listing of the element options making up a total transportation architecture for SEI missions. The options listed are all candidates for incorporation into architectures. Trade studies have not eliminated any of these options. (The list is representative and not necessarily complete.) The number of options on this chart for each row of options is indicated on the far right. In most cases, any option can be combined with any other set of options. Thus, the total possible combinations number in the millions. It is clear that available future effort can not hope to examine all combinations. This drives us to a strategy for architecture sensitivities analysis, to develop key trends and conclusions from relatively few architecture combinations.

Available Options

									No. of options
ETO	100 t.	140 t.	200+ t.	Add prop tanker					3 x 2
Node	SSF	Separate	SSF + separate	Self-assy.	Wet tanks	Refuel vehicles	Propellant depot		4 x 3
Lunar mode	Direct	Direct/ lunar ox.	LOR	LOR/ lunar ox.	L2/lunar oxygen				5
LTV	Cryo all-prop	Cryo aerobrake	NTR	NEP/SEP cargo	Fully reusable	Partially reusable	Expend- able		4 x 3
LEV	Cryo	Storable	Combined with LTV	Fully reusable	Partially reusable	Expendable			3 x 3
Mars mode	2.7 year	1.5 year							2
MTV	Cryo all-prop	Cryo aerobrake	NTR	NEP	SEP	Cycler			6
Mars node	LEO	L2							2
MEV	Cryo	Storable	Combined with LEV						3

Total possible combinations 2,799,360

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Top-Level Trade Table

The facing page considers mission profile, basing at Mars, and propulsion. Four important issues are central to mission profile selection: crew radiation exposure, crew time spent in zero g, the component of mission risk that increases with mission duration, and the added cost of shortening trip time. At one extreme is the notion, frequently expressed, that a Mars round-trip mission should be completed in a year or less. This is possible with certain advanced propulsion technologies, but at considerably higher cost than for longer trips, as described later in this section of the briefing. At the other extreme, trip time is seen as much less important than minimum mass and cost; conjunction profiles should be used. Crew time in zero g can be minimized by arrtificial-g spacecraft design. Increase in risk with duration is difficult to quantify. The mission duration issue presently is concerned mainly with cosmic ray exposure.

Crew radiation exposure comes from solar proton events (flares) and galactic cosmic rays, and from manmade sources if nuclear propulsion or power are used. Unshielded energy deposition from GCRs varies from 50 to 100 milligray (5 to 10 rad) per year. The low end of the unshielded range does not constrain Mars mission architectures, but the high end exceeds the present NCRP astronaut radiation guideline of 500 millisieverts/yr (this guideline is for space shuttle and space station missions; no guidelines have been given for Mars missions). It is possible that guidelines will be reduced in the future.

Five profile options are presented. Conjucntion fast transfer implies transfers much less than one year. Opposition/swingby trajectories vary from about 440 to about 550 days. Opposition/fast profiles imply 450 days or less, without swingby. The split sprint is a variation on the fast opposition profile in which the MEV and propellant for the return from Mars are sent in advance on a low-energy profile.

If galactic cosmic ray exposure must be controlled, we must either provide shielding on the transfer vehicle crew habitat or reduce exposure times. Shielding the transfer vehicle habitat dramatically increases its mass, requiring high performance propulsion such as nuclear, or favoring a cycler concept where massive habitats are emplaced on a suitable repeating trajectory and left there. To reduce exposure time, the applicable profiles are: (a) conjunction missions with fast transfers, i.e. less than 180 days, (b) fast opposition profiles, e.g. less than 1-year round trip, and (c) Mars surface rendezvous (Mars direct). The cycler/semi-cycler architectures offer shielding on the Earth-Mars leg, typically 5 months, and provides a 5-6 month conjunction transfer on the return trip. During the long stay at Mars, the crew must be on the surface most of the time unless a shielded Mars orbit habitat is also provided.

Fast-transfer conjunction missions may require orbit basing. A surface rendezvous mission may not be able to achieve the fast return transfer direct from Mars' surface with reasonable vehicle mass, because of the higher delta V required and because the payload launched from Mars' surface is the entire Earth return habitat rather than a lightweight, short-duration crew cab. Available propulsion options become very limited for fast missions. At one year, the only sensible options are NTR splits, where return propellant is prepositioned at Mars on a low-energy profile, or the use of a nuclear gas-core rocket. Below one year, the gas-core rocket quickly becomes the only option.

Top-Level Trade Table

Mission Profile	Propulsion				Basing	
	Cryo/ All-Prop	Cryo/ Aerobrake	NTR	NEP/ SEP	Orbit	Surface
Conjunction Minimum Energy	√	No advantage over propulsive capture	√	√	√	Later
Conjunction Fast Transfer	Excessive IMLEO	√	√	√	No. Reason for fast transfer is less GCR dose	√
Opposition/ Swingby	Same	√	√	Note 1	√	As a resupply mode
Opposition/ Fast	Same	Excessive IMLEO	√	Not able to make fast trips	√	Same
Opposition/ Split Sprint	Same	Same	√	Cargo only	√	Same

Note 1: NEP flies an opposition/swingby-like-profile but does not benefit from Venus swingby.

Architecture Results for Three Activity Levels

The top-level architecture selection results for the three activity levels are shown on the facing page. For the minimum program, a cryogenic expendable tandem-staged direct mode is the clear economic winner. Its lower development expense causes the operational cost savings for a reusable LOR system to have little payoff. At the median activity level, the reusable system gives about a 5% return on investment (ROI). Our baseline program included lunar oxygen at the median level, but the ROI is estimated only about 3%. At the high lunar activity level, reusable systems and lunar oxygen both have strong payoff, e.g. the lunar oxygen ROI is about 10%

The minimum Mars program is most economic with cryogenic all- propulsive expendable vehicles on conjunction profiles. The NTR has an ROI less than 2% at this level. If natural environment radiation concerns lead to a conjunction fast transfer or opposition profile, the NTR is the preferred solution with cryogenic/aerobraking as a backup. At the median level, the NTR has a 16% ROI versus cryo all-propulsive. Here also, aerobraking is a backup and SEP comes into the picture as a "dark horse", with about 10% ROI if array costs can be reduced to $100/watt, a tenfold reduction from present costs. At $500/watt, the SEP has a negative 10% ROI, showing the great leverage of array cost. At the high level, electric propulsion is indicated as important, but development costs are a problem unless low-cost SEP arrays can be produced. If electric propulsion costs are too high for a settlement-scale Mars program, the NTR/dash and Mars direct modes are viable options.

Architecture Results for Three Activity Levels

Minimum

Lunar:

Expendable

Mars:

- Cryogenic all-propulsive
- Unless radiation environment requires reduced trip times; then nuclear rocket or cryo aerobrake conjunction fast transfer

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Median (full science)

Lunar:

Start expendable, possible growth to LOR reusable, aerobraking

Mars:

- Nuclear rocket, conjunction, multiple landers
- Opposition or conjunction fast transfer options
- Cryo/aerobraking backup
- SEP "dark horse"

Industrialization /settlement

Lunar:

LOR crew and tandem direct cargo, reusable, with lunar oxygen

Mars:

- Early cryo/all-propulsive option
- Electric propulsion for sustained growth (probably SEP)
- Nuclear rocket/dash or Mars direct/Mars propellant, options for crew rotation and resupply.

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Seven Architecture Recommendations

The next seven pages contain our main architecture recommendations with data illustrating key points.

Lunar Architecture

- <u>Begin the lunar program with a tandem-direct expendable system.</u>
  - System can be designed to eliminate on-orbit assembly; one docking or berthing required.
  - The number of development projects is minimized. Offers reasonable expectation of return to the Moon by 2004 under likely funding constraints.
  - Flight mechanics constraints for LOR operations are avoided.
  - Tandem-direct LTV is a starting point for evolution to all other identified lunar architectures.
  - Lunar aerobrake can be tested on the unmanned booster stage without risk to the crew. Stage is otherwise expended.

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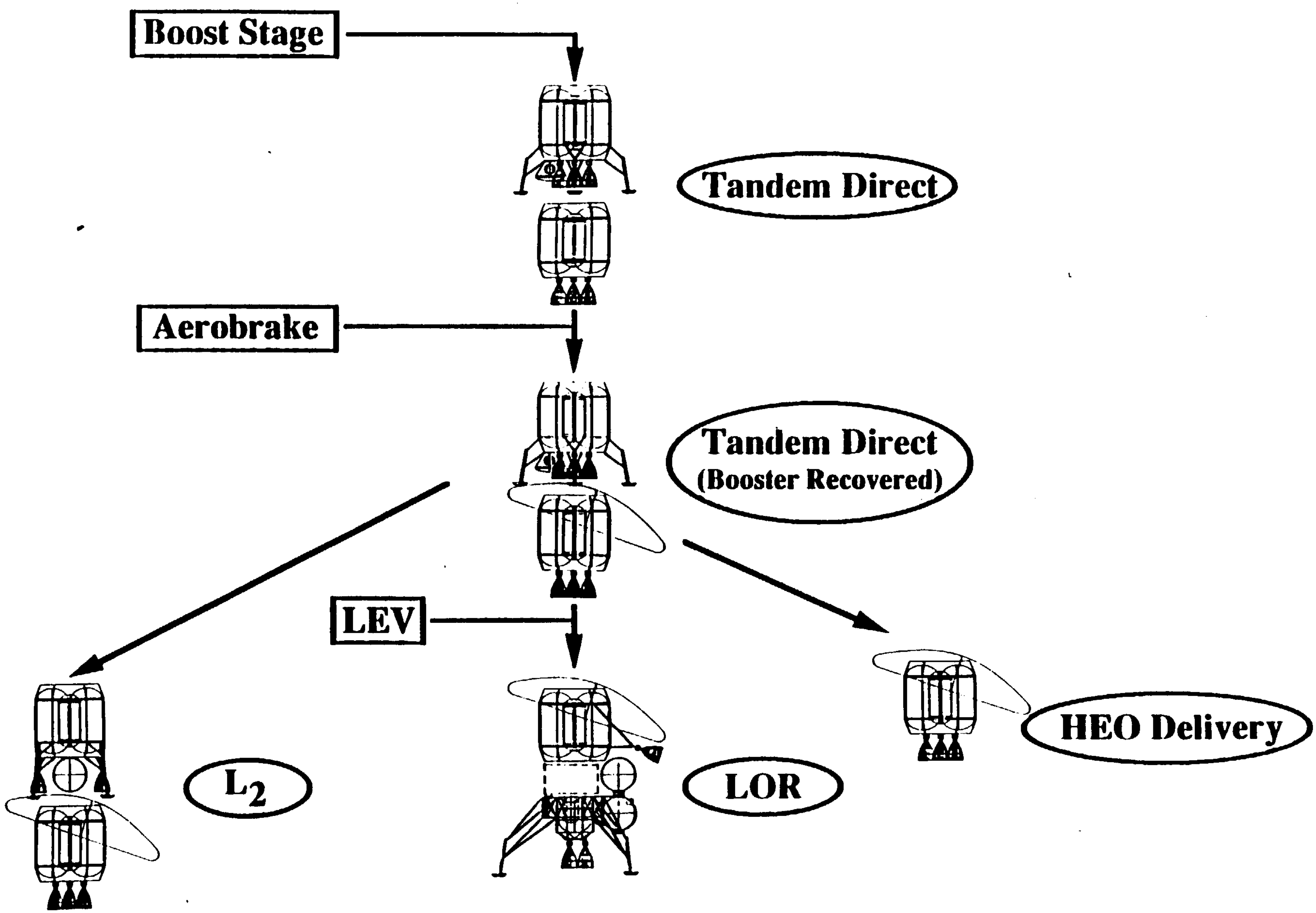
Lunar Transportation Family
Evolution



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Lunar Cryogenic Propulsion

- <u>Invest in cryogenic storage and management technology.</u>
    - Without advanced development of a low-boiloff flight-weight cryogenic insulation system, the lunar program may be forced to a storable propulsion system for lunar vicinity operations. Cost impact is billions of dollars.
- <u>Invest in a 30K-class advanced expander cryogenic engine with 10:1 or better throttling capability.</u>
    - An advanced expander engine offers about 20 seconds' Isp gain over a modified RL-10; can demonstrate advanced health monitoring and maintainability features essential for Mars missions.

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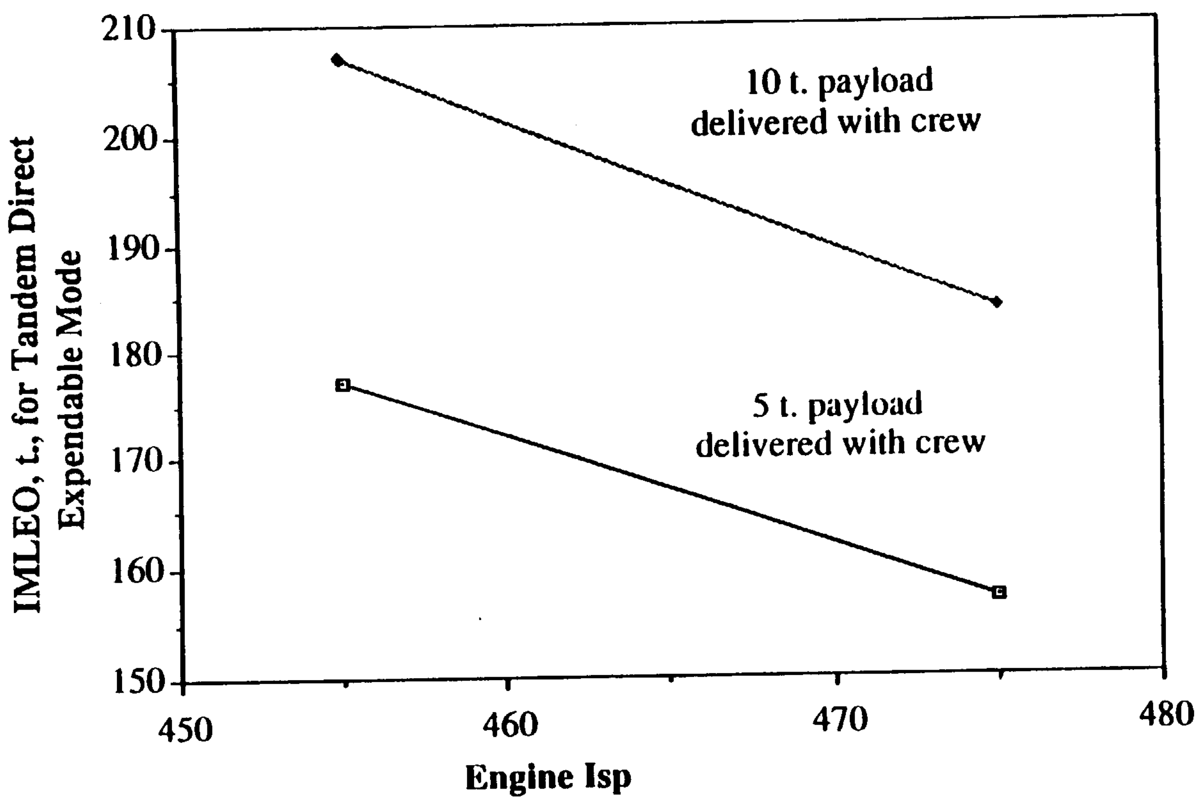
Early Lunar Mode Sensitivity to Isp



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Mars Baseline Architecture

- Baseline nuclear thermal rocket propulsion for Mars.
  - Nuclear thermal rocket indicated as very economic and flexible over wide range of program activity levels.
  - Nuclear rocket vehicle mass is sensitive to specific impulse. Isp gain for carbide fuels is well worth the technology investment.
  - Development and qualification testing requires proven test facility technology that contains hydrogen effluent and scrubs radioactivity.
  - Nuclear rocket performance permits modest lunar program and significant Mars exploration with about six launches per year of 100-tonne class HLLV.
  - Nuclear rocket baseline offers reasonable expectation of initial Mars mission by 2010 under likely funding constraints.

- Recommended technology advancement program:
  - High-performance fuels
  - Full-containment ground test facilities.

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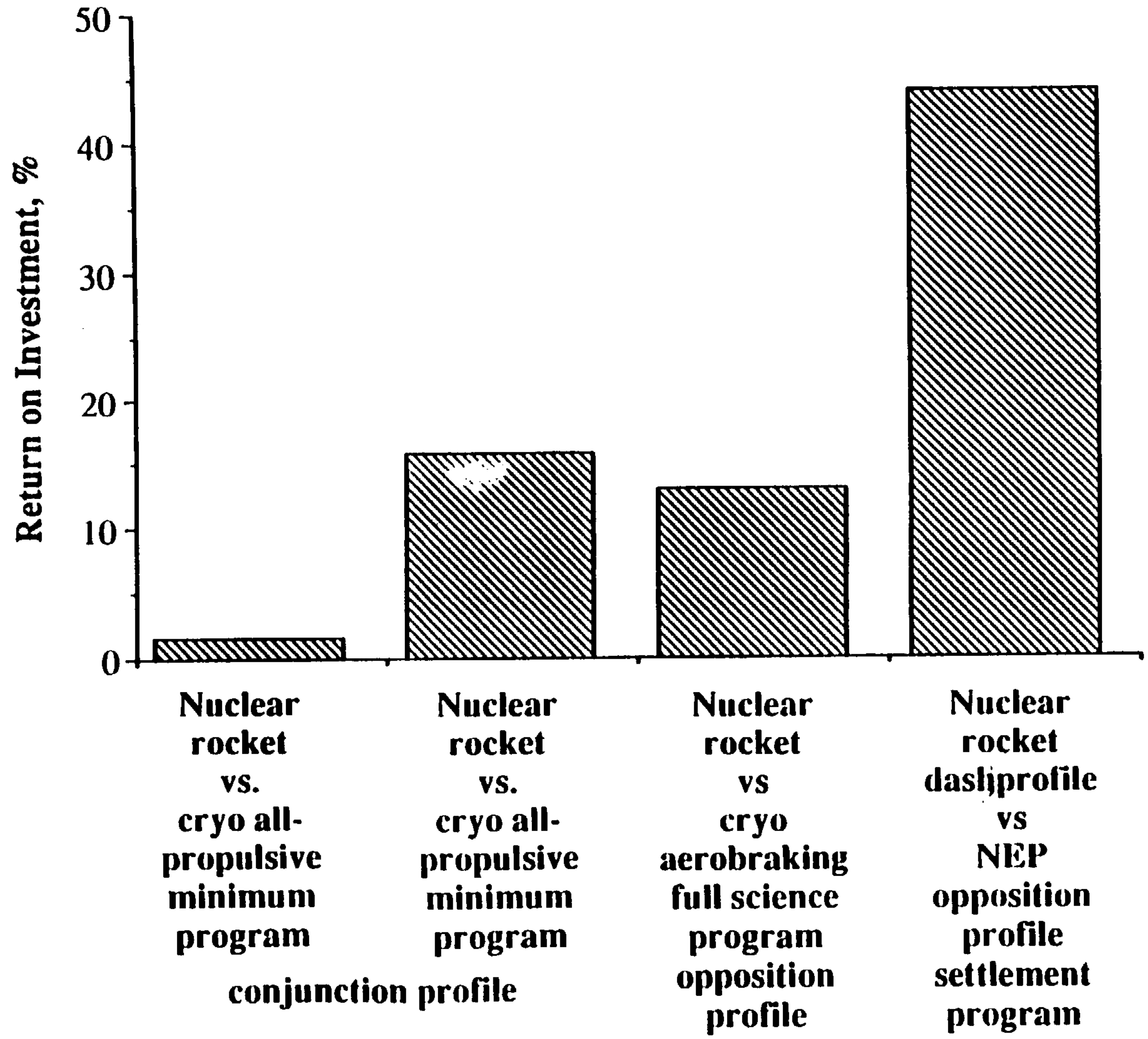
Nuclear Rocket ROI Trades



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Aerobraking Technology

- <u>Accelerate aerobraking technology for Mars aerocapture as backup to nuclear rocket.</u>
  - Target decision between the two in the 1996-2000 time frame.
  - NTR performance and cost uncertainties, especially test facilities and testing, merit backup.
  - Aerobraking needed for Mars landing. Technology challenges less daunting than aerocapture, but merit technology program.
  - Aerobraking technology keeps other options open.
    - Conjunction fast transfer
    - Mars direct
    - Cycler orbits
    - NTR-dash profile
  - Aerobraking is economic for lunar transportation at >= two flights/year.

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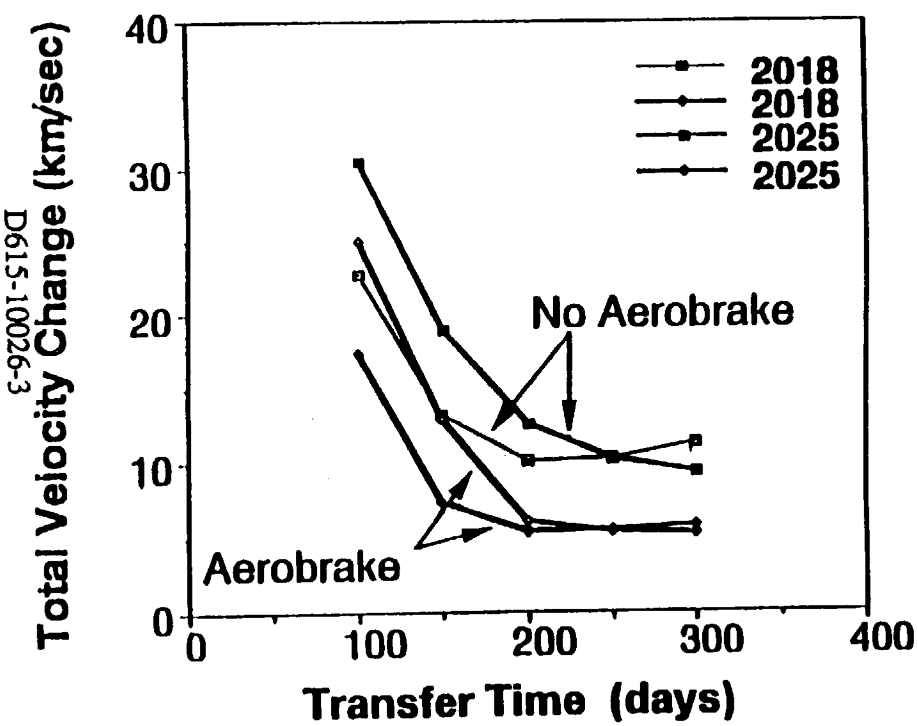
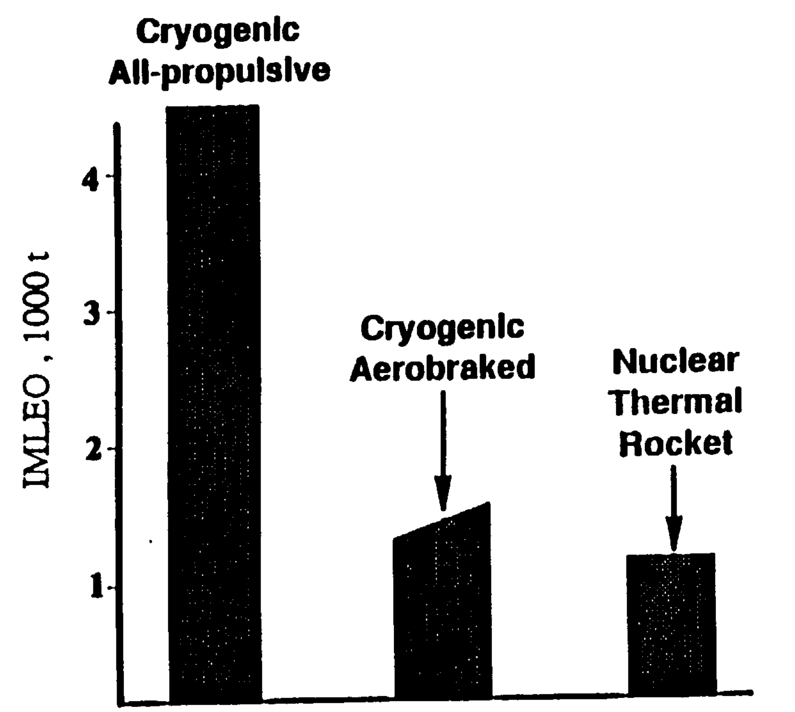
Mars Aerocapture Benefit for Conjunction Fast Transfer





Conjunction Fast Transfer

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Program Implementation Architectures Relation to Aerobraking

The facing page indicates uses of aerobraking for the various architectures. As noted, some form of aerobraking occurs in all of the architectures, in particular for Mars landing and for Earth capture on return from lunar missions. In addition, some of the architectures include an Earth crew capture vehicle (ECCV) for direct return of the crew to Earth in cases where, for example, an NEP or SEP vehicle must spiral back down to LEO or in the case of an NTR where the vehicle captures into a highly elliptic orbit.

Program Implementation Architectures

Architecture	Features	Aerobraking Functio: Mars cap	Mars land	Earth cap/ lunar	Earth cap/ Mars	Earth entry*
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	x	x	x	x	x
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.		x	x		x
L2 Based cryogenic/ aerobraking	L2-based operations; optional use of lunar oxygen.	**	x	x	x	x
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	x	x	x	x	
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	***	x	x	x	x

Notes: * optional/emergency mode **opposition class only *** MEV-class crew taxi (not a large MTV)

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Aerobraking Flight Test Bed

- <u>Perform aerobrake tests on the LTV booster, to put the technology on the shelf for Mars.</u>

  - If the lunar program grows to high activity levels, lunar aerobrake is economically justified.
  - A space-assembled aerobrake is needed for Mars landing.
  - Aerocapture technology is needed as backup to Mars NTR.

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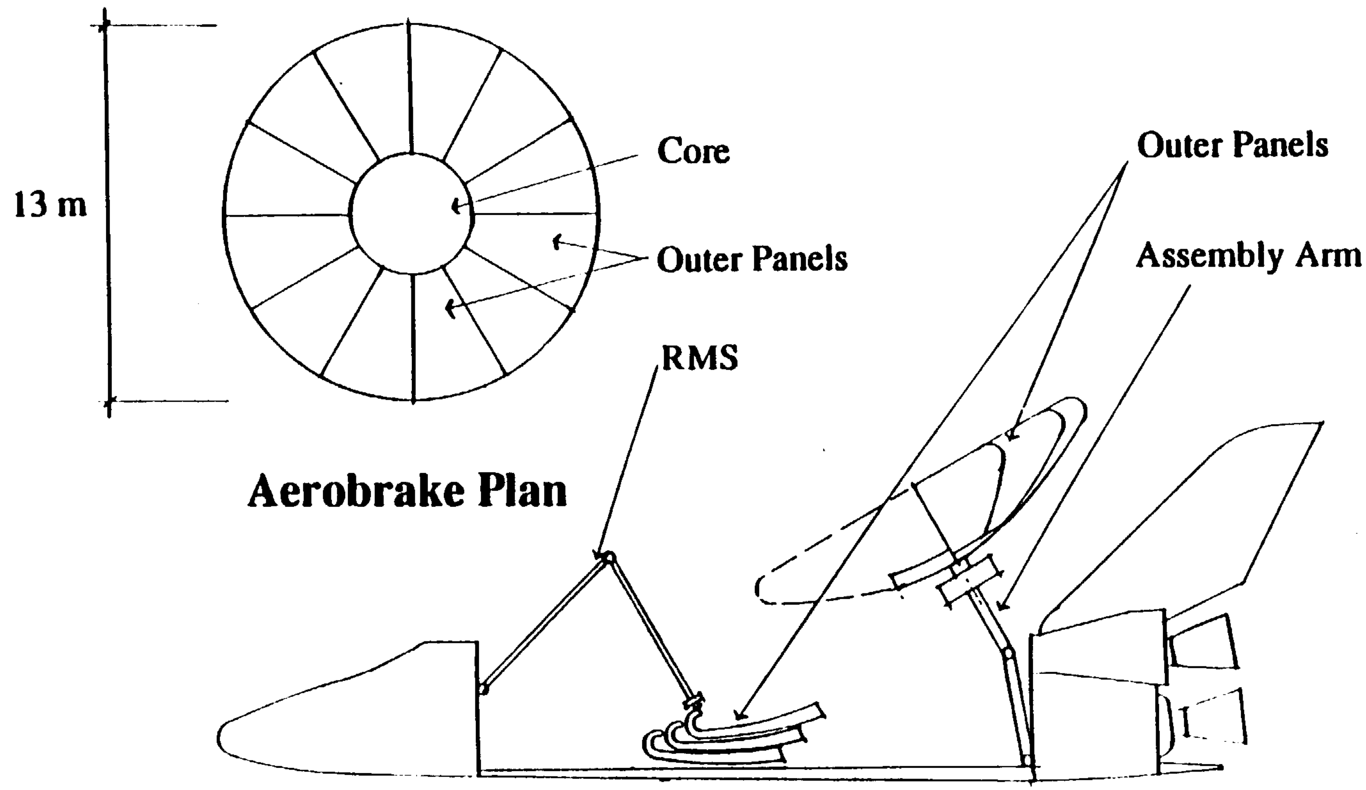
Aerobrake Assembly Test in LEO



Aerobrake Plan

Cross Section

Assembly arm rotates brake as outer panels are installed for easy RMS reach and crew visual contact during operations

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SEP as "Dark Horse"

- <u>Designate solar-electric propulsion (SEP) as a "dark horse" for Mars transportation.</u>

  - Technology advancement issues:

    - Light weight, high performance, radiation resistant arrays.
    - Automated production technology, $100/watt
    - Robotics technology for constructing SEP and deploying arrays
    - Long-life, high power density, efficient electric thrusters

  - If safety precludes operation of nuclear propulsion in low Earth orbit, SEP is the only option more economic than cryo-genic/aerobraking.

  - If low-cost array target achieved, SEP is more economic than NEP.

  - SEP is the most likely architecture for eventual private sector use for Mars settlement.

  - SEP technology has derivative benefits, e.g. power beaming to planet surfaces.

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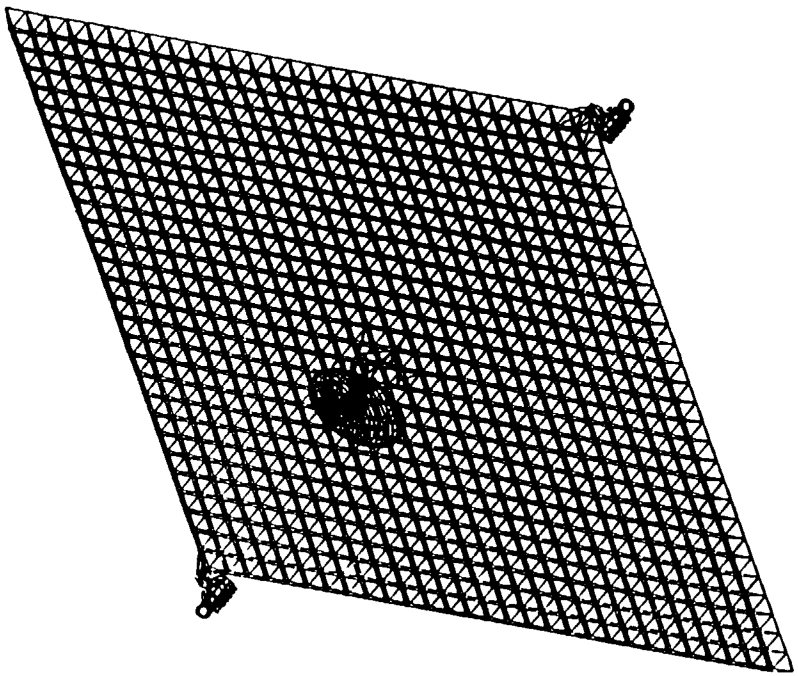
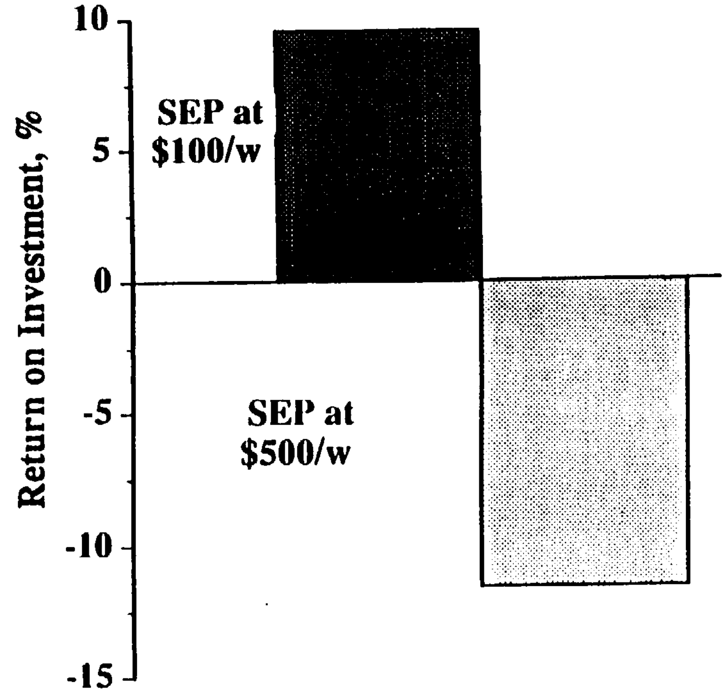
Return on Investment Comparison
SEP versus NTR; Full Science Program





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Nuclear Space Power

- Continue the nuclear space power program towards near-term systems applicable to planet surface power.

  - DDT&E and production cost estimates from this study eliminate nuclear electric propulsion (NEP) as a top contender, but are very preliminary.

  - As NEP systems are better understood, estimates may come down.

  - To keep NEP option open:

    - Further studies to better understand the cost of nuclear power systems suitable for electric propulsion.
    - Modest funding of high-leverage high-performance power conversion technology.

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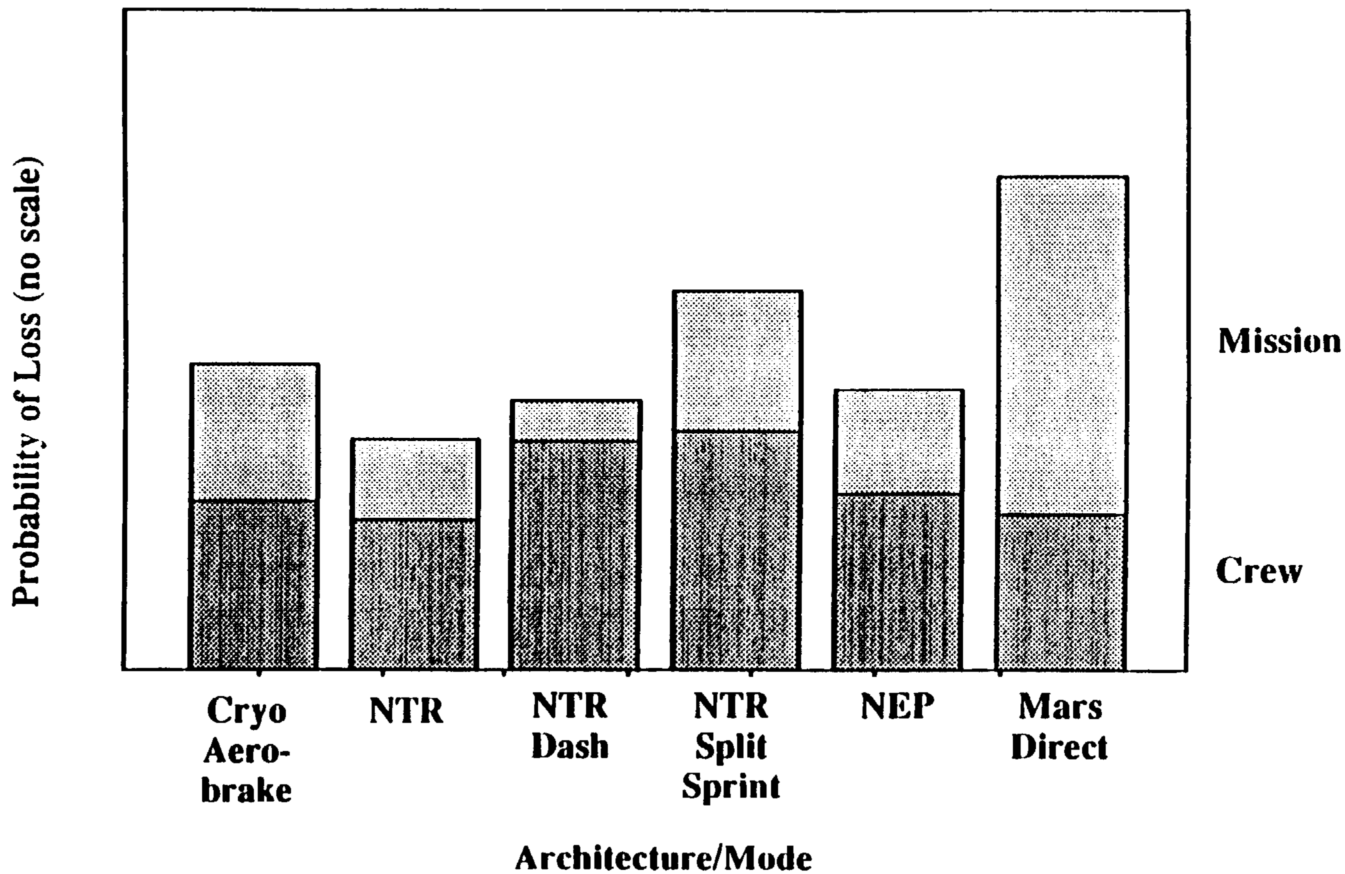
Mission Risk Comparison

Mission risks were compared in a semi-quantitative way. The methodology is rigorous and quantitative, but reliability and safety estimates for SEI hardware and maneuvers are no more than ballpark guesses today. We made representative estimates with an attempt to be consistent, i.e. the same type of maneuver was given the same number for all cases. Plausible differences were used, e.g. aerocapture was judged higher risk than propulsive capture. Abort modes were included where available.

The facing page shows comparative risks for crew loss and mission loss for several architectures and modes.

NTR shows the least risk because of the propulsive capture advantage, and because a free return abort was assumed, as it was for the cryo/aerobrake. The NTR/dash mode does not permit free return abort or descent abort at Mars, so some mission loss risk turns into crew loss risk. As Mars transportation matures and a safe refuge on the surface of Mars is available, the NTR/dash mode is deemed acceptable. The NTR split sprint mode also exhibits higher risk because of lack of abort modes, e.g. no free return. NEP is shown comparable to, but slightly riskier than NTR. The NEP case is sensitive to the lifetime dependability of the propulsion system; this figure is much more uncertain than NTR reliability. Mars direct has a higher mission loss risk because of its complex automated operations, but the crew loss risk is comparable to the others. The perception of crew loss risk for Mars direct is probably higher than the real risk.

Mission Risk Comparison



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Man Rating Requirements

The facing page describes our recommended approach to man-rating and lists the systems/subsystems for which we believe man-rating is required.

Man-Rating Requirements

Approach

- Ground-based testing wherever possible.
- Use flight program activities to bootstrap, e.g. lunar aerobrake program builds confidence in Mars aerobrakes.
- Flight demonstration of critical functions, e.g. Mars cargo landing, before critical manned use.
- Life demo for long-duration systems before critical manned use, e.g. ECLSS on SSF or lunar surface before manned Mars mission.

Subjects

- Aerobrakes
- Cryogenic rocket engines
- Nuclear rocket engines
- Cryogenic propellant systems
- Attitude control propulsion systems
- Nuclear & solar electric propulsion systems
- ECLSS/TCS
- Crew modules/hab systems
- Vehicle power
- Avionics & Communications systems
- Surface transportation systems

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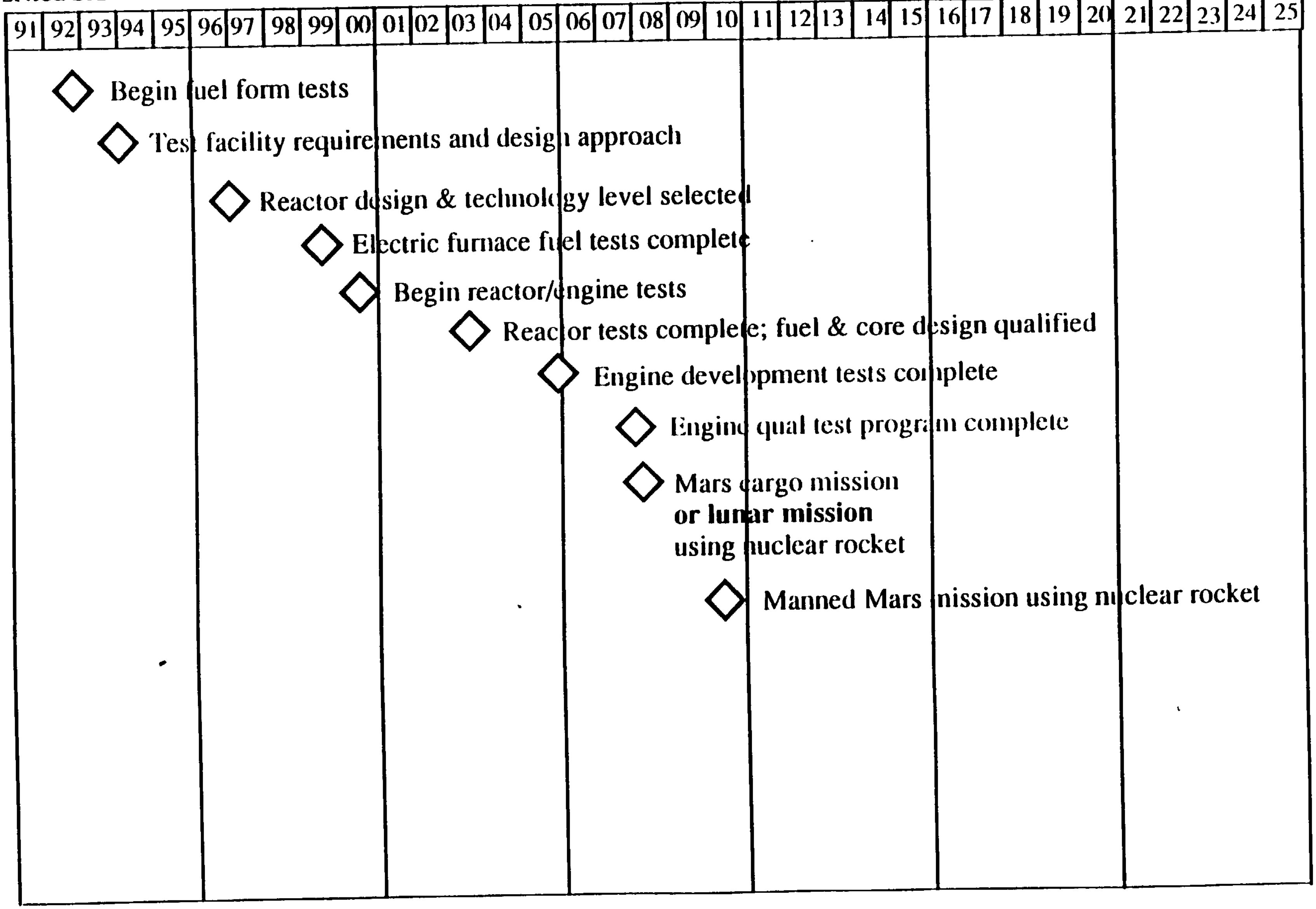
Nuclear Rocket Man-Rating Approach

A sequence of major tests and demonstrations to achieve nuclear rocket man-rating is shown. Note that two flight demonstration options exist. A decision of which to use depends on whether cargo delivery to Mars is needed before the first manned mission, as would be the case if a conjunction fast transfer and long surface stay is required on the first mission to reduce galactic cosmic ray exposure to the crew.

Nuclear Rocket Man-Rating Approach



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Technology Advancement and Advanced Development

The next three charts present our current recommendations for technology advancement and advanced development, with schedules and funding estimates. The funding level averages about $300 million per year. If we consider the median (full science) program as representative, the technology/advanced development program is about 0.2% of the life cycle cost of the program to 2025, a very modest investment.

ADVANCED CIVIL SPACE SYSTEMS

Technology Development Schedules
- Overview -

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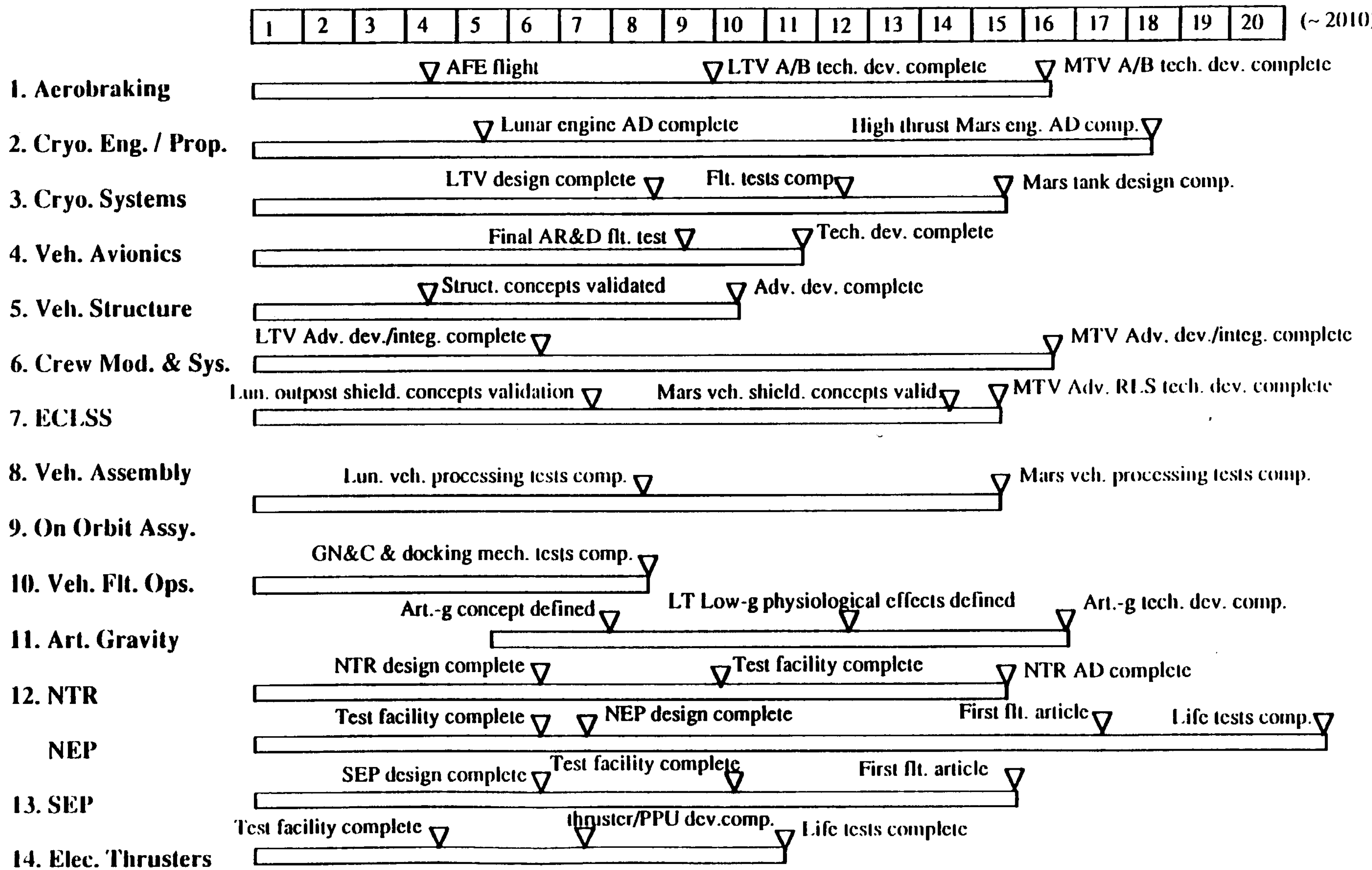

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Technology / Advanced Development Funding Estimates

Technology Category	1	2	3	4	5	6	7	8	9	10	11	Total
1 - Aerobraking* -Technol.	1	6	10	5	5	8	10	10	8	5		68 M
- Adv. Dev.	0	0	30	30	55	20	30	65	65	40	30	400 M
2 - Cryogenic Engines / Prop.	0	30	30	30	20							110 M
- Adv. Dev.	0	0	99	71	65	50	65	65	50			465 M
3 - Cryogenic Systems - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	10	10	20	50	50	50	110				300 M
4 - Vehicle Avionics/Software	2	5	5	5								17 M
- Adv. Dev.	0	0	0	25	45	40	40	40	40	40		270 M
5 - Vehicle Structures - Tech.	3	7	5	5	7	7	5					39 M
- Adv. Dev.	0	0	15	17	11	10	15	15	15	10		108 M
6 - Crew Modules & Systems	0	0	3	3	3	5	5	5	3			27 M
- Adv. Dev.	0	0	15	20	10	10	15	20	20	10		120 M
7 - Environ. Ctrl. & Life Supp.	0	0	3	5	5	10	10	5	5			43 M
- Adv. Dev.	0	0	6	10	10	15	30	30	30	20		151 M

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ADVANCED CIVIL SPACE SYSTEMS

Technology / Advanced Development Funding Estimates

BOEING

Technology Category	1	2	3	4	5	6	7	8	9	10	11	Total
8 - Vehicle Assembly - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	5	40	40	40	40	40	40	10			255 M
9 - Orbit Launch & Checkout	5	5	5	5								20 M
- Adv. Dev.	0	4	15	16	5	10	10	10	10	5		85 M
10 - Vehicle Flight Operations - Adv. Dev.	0	0	9	15	10	15	15	15	10	5		94 M
11 - Artificial Gravity - Tech. - Adv. Dev.	0	0	0	2	5	10	10	10	10	3		50 M
12 - Nuclear Propulsion												
NTP -	0	10	15	20	20	20						85 M
NEP -	0	15	20	30	30	30	20	20				165 M
13 - Solar Electric Ion Prop.	2	8	10	15	15	10						60 M
Array manufac. Tech. -	0	0	0	30	30	30						90 M
14 - Electric Thrusters	0	5	10	20	20	20	10					85 M
Tech. Development Total	**23**	**120**	**367**	**482**	**461**	**410**	**380**	**460**	**276**	**138**	**30**	**3147 M**

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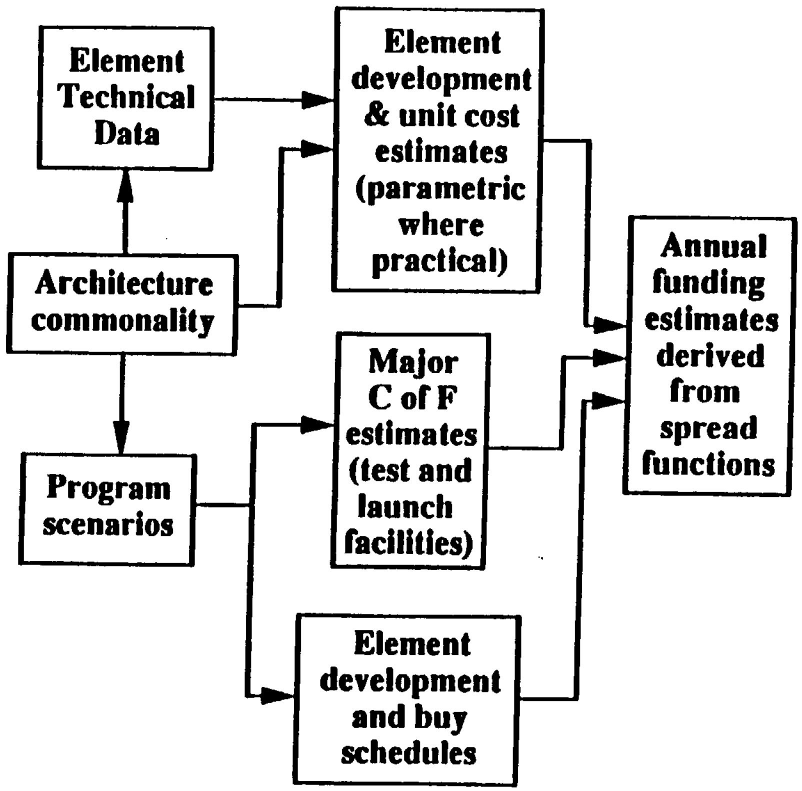
Life Cycle Cost Model Approach

Our basic cost model kernels are parametric cost models. We use the Boeing Parametric Cost Model and the RCA Price models to estimate development and unit cost. The determination of hardware to be costed comes from what architectural elements are needed and from element commonality of the architecture. Program schedules determine requirements and timing for major facilities and for the element development and buy schedules. All of these inputs are used to estimate annual funding for each component of the program, using cost spread functions. The costs are integrated into a spread sheet life cycle cost model to obtain annual funding for complete programs.

The ground rules used in this analysis are indicated on the chart.

The ground rule for use of closed ecological life support (CELSS) and lunar oxygen comes from economics trade studies conducted several years ago through last year.

Life Cycle Cost Model Approach



Ground Rules

- No precursor missions costed.
- NASA contingency not added
- Common element in new application gets 25% delta DDT&E cost.
- No production learning unless production rate > 1 per year.
- Production rates maintained minimum of 1 per 5 years to keep lines open.
- Mission definitions flexible to enable transportation systems to operate at high efficiency.
- All scenarios include closed ecological life support and ISRU for efficiency.

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Architectural Cost Drivers

Our investigations of architectures, while preliminary, indicate the importance of cost drivers, in the order listed on the chart. The number of development projects should be minimized through commonality and phased by evolution so that development costs are reduced and are spread over the life cycle of the program, rather than lumped early in the program.

Space hardware for SEI missions is expensive and should be reused if possible. As an example, our unit cost estimate for the Mars transfer crew module is more than a billion dollars. Reuse of this equipment motivates investment in the advanced transportation technology needed to make it reusable.

The third point is that Earth launch mass drives Earth launch cost. Even if Earth launch cost is reduced by ALS-class vehicles, the Earth launch cost is the largest single part of program cost.

The final point is that design and development of systems with mission and operation flexibility enhances commonality and minimizes the risk that changes in mission requirements force new developments or major changes.

Architecture Cost Drivers

- Number of development projects (minimize through commonality)
- System reuse (maximize)
- Earth launch mass (minimize)
- Mission and operational flexibility (maximize)

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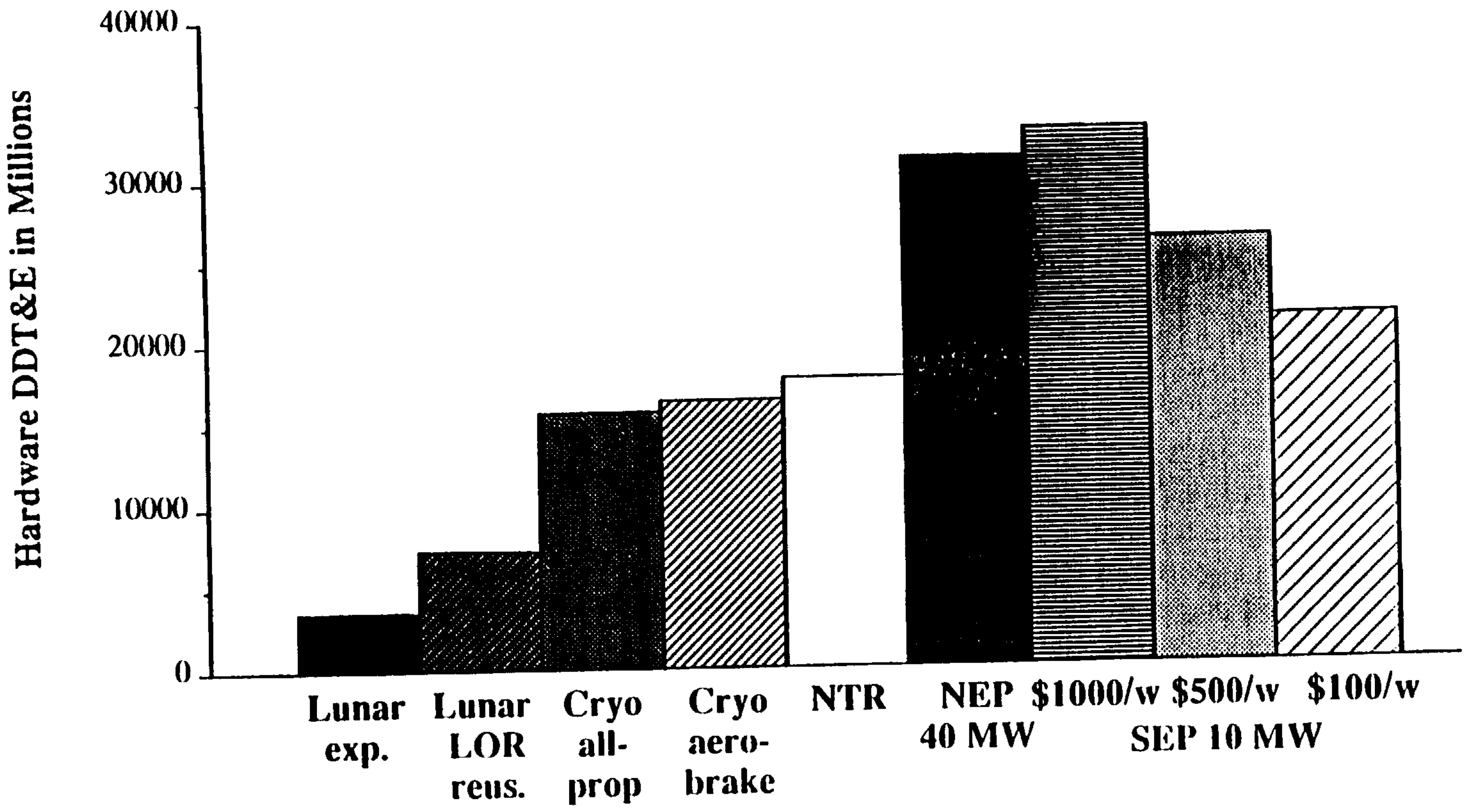
In-Space Transportation DDT&E Comparison



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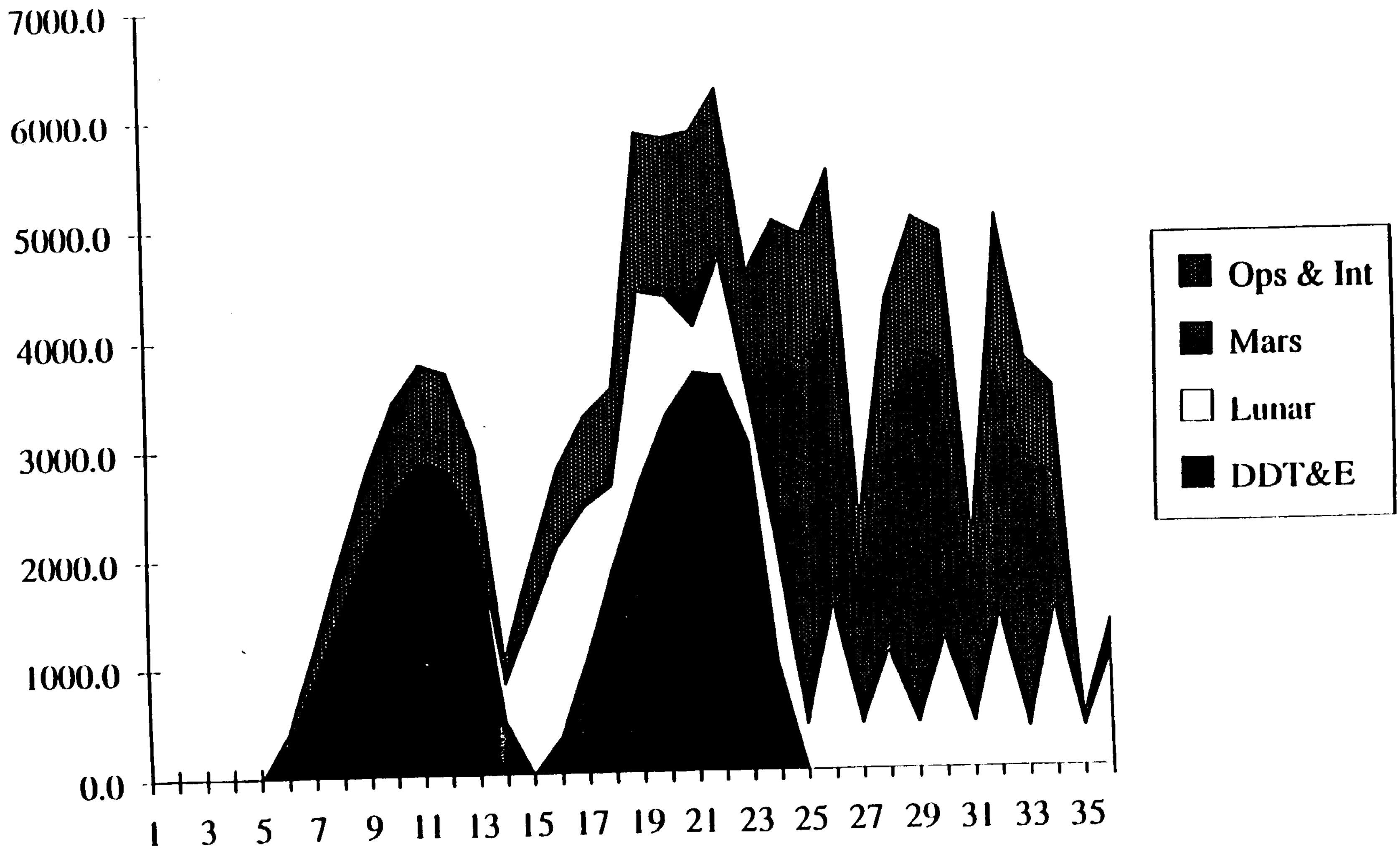
Minimum Program Life Cycle Cost Spread

The minimum program life cycle cost spread peaks between five and six billions per year. The deep valley between lunar and Mars peaks indicates that the Mars program should occur earlier in this program. The minimum program involves relatively modest investments in surface systems and falls well below the SEI funding wedge implied by the Augustine Committee recommendations.

Minimum (Baseline W/Ops & Int)



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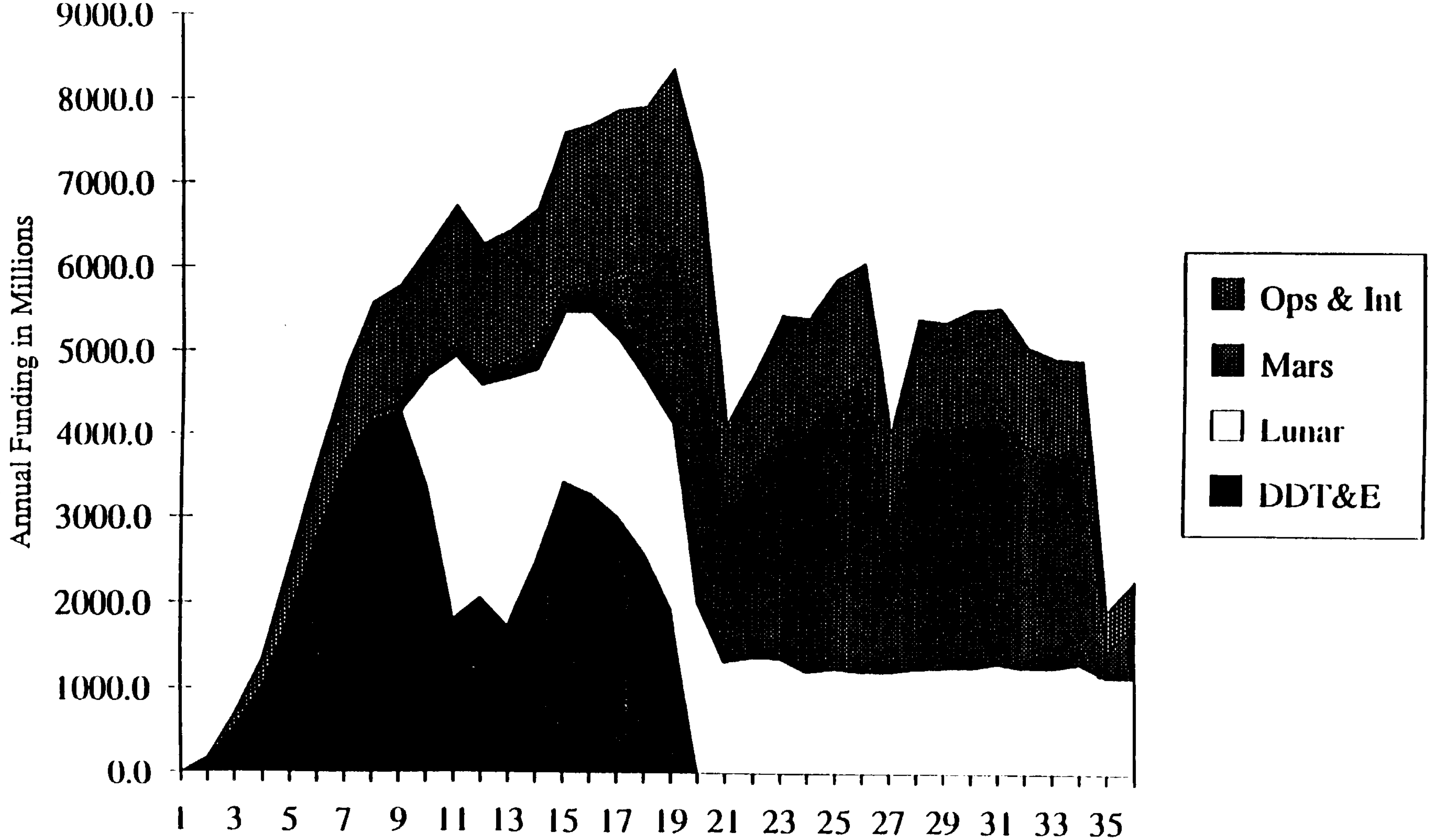
Median (Full Science) Program Life Cycle Cost Spread

The median life cycle cost spread peaks at about eight billions per year. With addition of likely surface systems costs, this program probably exceeds the Augustine guidelines during the peak years.

The median program exceeds by a factor of several the science and exploration potential of the minimum program. Lunar human presence grows from an occasional 45 days to permanent presence of six people, and Mars surface time grows from about four man- years to about 30. In other words, a roughly 50% increase in cost leads to about an order of magnitude increase in exploration and science potential.

Full Science (Baseline W/Ops Int)



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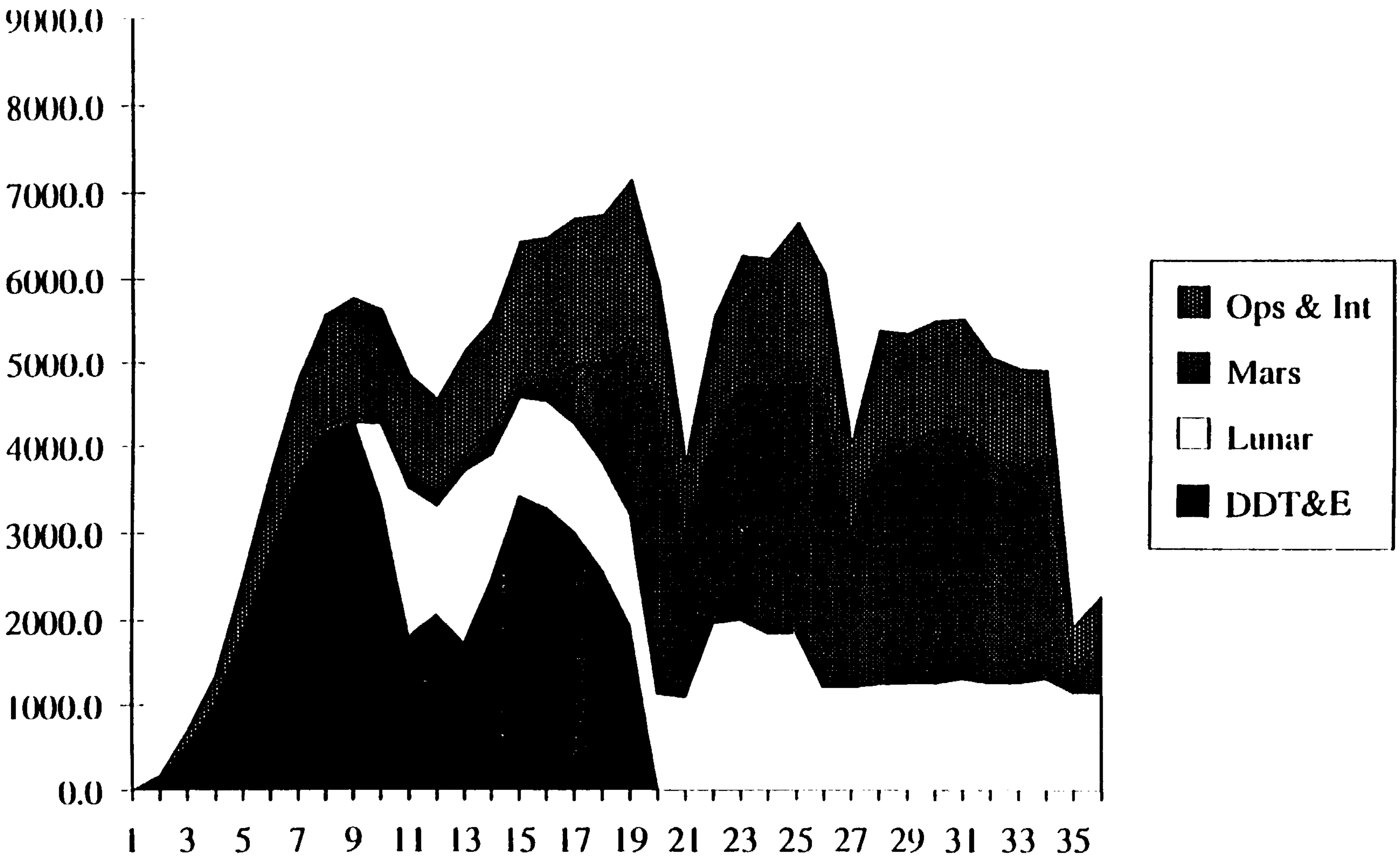
Median (Full Science) Program Life Cycle Cost Spread
Reduced Early Lunar Program

By deferring major lunar activities, the median program can be brought within the Augustine guidelines. Permanent human lunar presence is delayed until after the Mars DDT&E peak. The early lunar program is like the minimum scenario, i.e. man-tended astrophysics observatories.

Another way to level the funding profile for the median program is to defer Mars by a few years. The reference median program achieves a Mars landing in 2010 (2009 departure). Deferral to about 2016 would probably smooth out the funding profile much as did the reduction of the early lunar program.

Our view was that getting to Mars early was more important than an early buildup to permanent lunar presence. The partially deferred lunar program represented here still achieves astrophysical observatories early, but defers permanent human presence until after the major Mars mission DDT&E is complete.

Full Science (Baseline W/Ops Int-Reduced Lunar)



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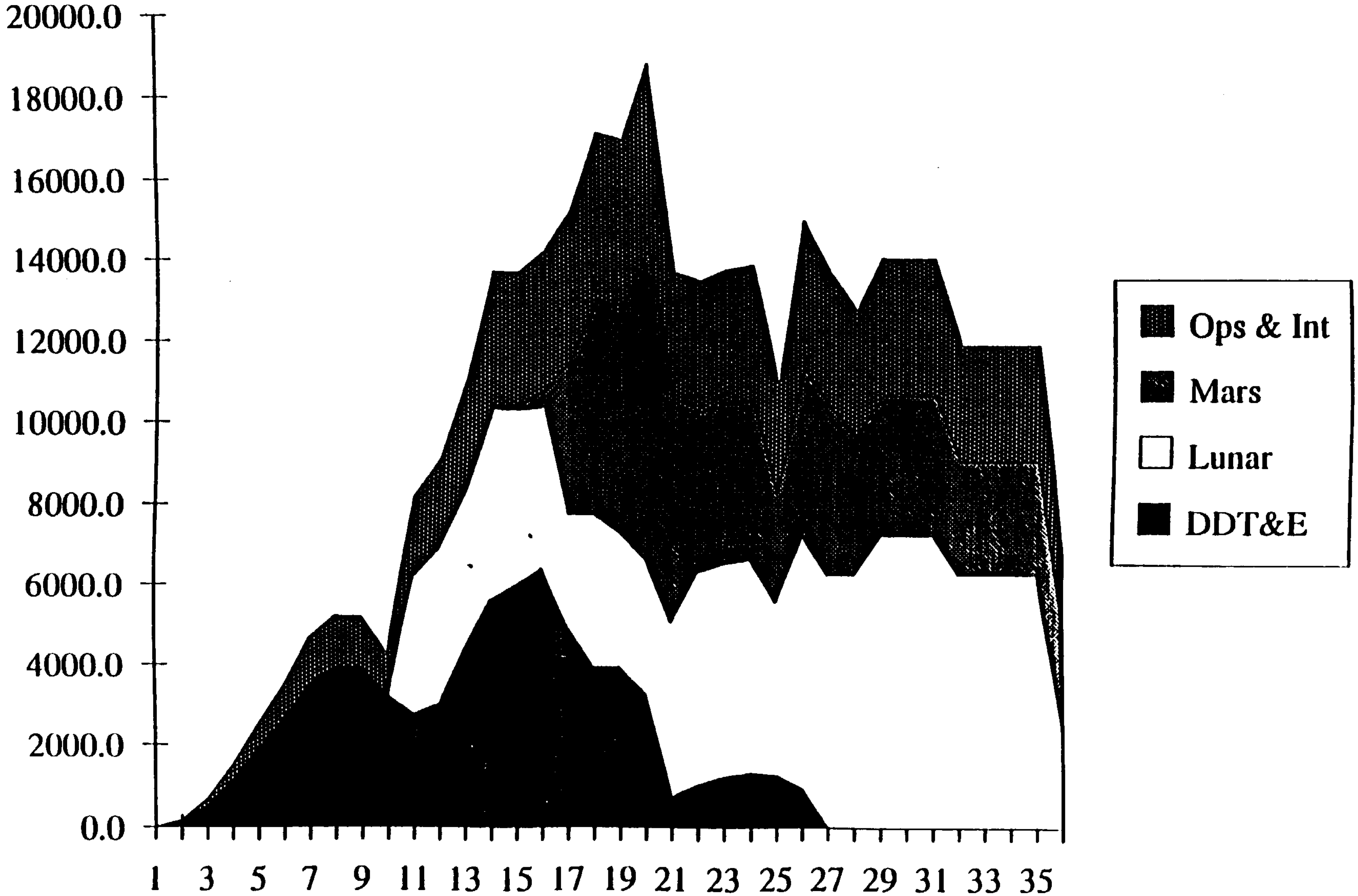
Industrialization and Settlement Cost Spreads

Our maximum scenario involved simultaneous industrialization of the Moon and progress towards settlement of Mars. As the cost spread shows, this is clearly beyond the funding levels recommended by the Augustine Commission. Both of the premises of this scenario, however, suggest significant private sector involvement.

What is significant in the result presented here is that investment on the order of $100 billions over about 20 years stretches from a plausible public-sector program of science and exploration to a program also involving the private sector for industrialization and settlement. This amount of funding is more than the private sector investment in the Alaska oil pipeline by a factor of a few, and probably less than the private investment in oil supertankers since the closure of the Suez Canal.

The economic potentials of lunar and/or Mars industrialization and settlement are presently not at all understood. We have made some stabs at estimating the costs. We have little or no idea as to the eventual payoffs.

Settlement/Industrialization Baseline w/Ops Int



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Results of Return on Investment Analyses

The facing page summarizes results of return on investment analyses. (The ROI methodology is explained in the technology and programmatics section of this briefing book.) Results designated "no ROI" had one case always more expensive than the other. An ROI can be calculated only when funding streams cross.

The storable case has very negative ROI because while less (i.e. no) technology money is spent, more vehicle stages must be developed so that the negative cost impact of not doing the essential cryo management and engine technology is large and early. The case for reusable lunar transportation is negative for a minimum lunar program and weak for a median program; it is strong for an industrialization-class program.

The other results were discussed earlier and are included here for completeness.

Return on Investment Analysis Summary

Case	Stor LOR vs cryo direct exp	Reus. LOR vs cryo direct exp	Reus. MEV vs exp MEV	SEP vs NTR $100/w	SEP vs NTR $500/w	SEP vs NEP	NTR vs cryo all-prop	NTR vs cryo all-prop	NTR vs cryo aero brake	NTR dash vs NEP	Lunar oxygen	Lunar oxygen
Program	Min	Full science	Ind/ settl	Full science	Full science	Any	Min	Full science	Full science	Ind/ settl	Full science	Ind/ settl
Result	-85	4.9	No ROI	9.6	-11	No ROI	1.7	15.9	13	44	4	10
Conclusion	Cryo	Reuse case weak	Reus MEV higher LCC	SEP	NTR	SEP better if less cost	CAP	NTR	NTR	NTR	No LLOX	LLOX

Strategy for Architecture Synthesis

The strategy we have adopted is illustrated on the facing page. First, we examined propulsion systems options through trade studies to understand how they work and to define preferred configuration operating modes. Secondly, based on the knowledge gained through these trade studies we chose a set of architectures using combinations of systems and modes, paying attention to integration compatibility, evolutions and commonality. Third, we will compare and trade architectures over a range of scopes and obtain important sensitivities and understand how architectures respond to program scope. We expect this analysis to lead to preferred architectures for various scopes. The final step is to conduct trades within the winning architectures to make further improvements.

All of this is guided by knowledge of the architecture cost drivers described earlier and by the knowledge gained on how systems work together, from the trades conducted within individual propulsion systems.

Strategy for Architecture Synthesis

Conduct trades on systems to find preferred configurations and operating modes → Synthesize candidate architectures using effective combinations of systems and modes → Compare and trade architectures over a range of program scopes to obtain important trends and sensitivities → Conduct trades within winning architectures to improve them

Criteria:	Criteria:	Criteria:	Criteria:
Risk	Integration compatibility	Ability to satisfy program goals & schedules	Risk
Performance	Evolution	Cost	Performance
Cost	Commonality of hardware & technology		Cost
Flexibility			Flexibility

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Architectures Synthesis vs Mission/System Analysis

The facing page compares this approach to the traditional top-down systems engineering approach. The traditional approach shown on the right, starts with program goals, establishes mission requirements through trades, and continues to lower levels. As usually conducted, the traditional approach is faced with the great number of possible combinations noted earlier. The usual outcome is that requirement decisions are made and systems selected without trade studies.

The synthesis technique, on the left, attempts to avoid this problem by a combined top down/bottom up approach. It is similar to a classical optimization problem.

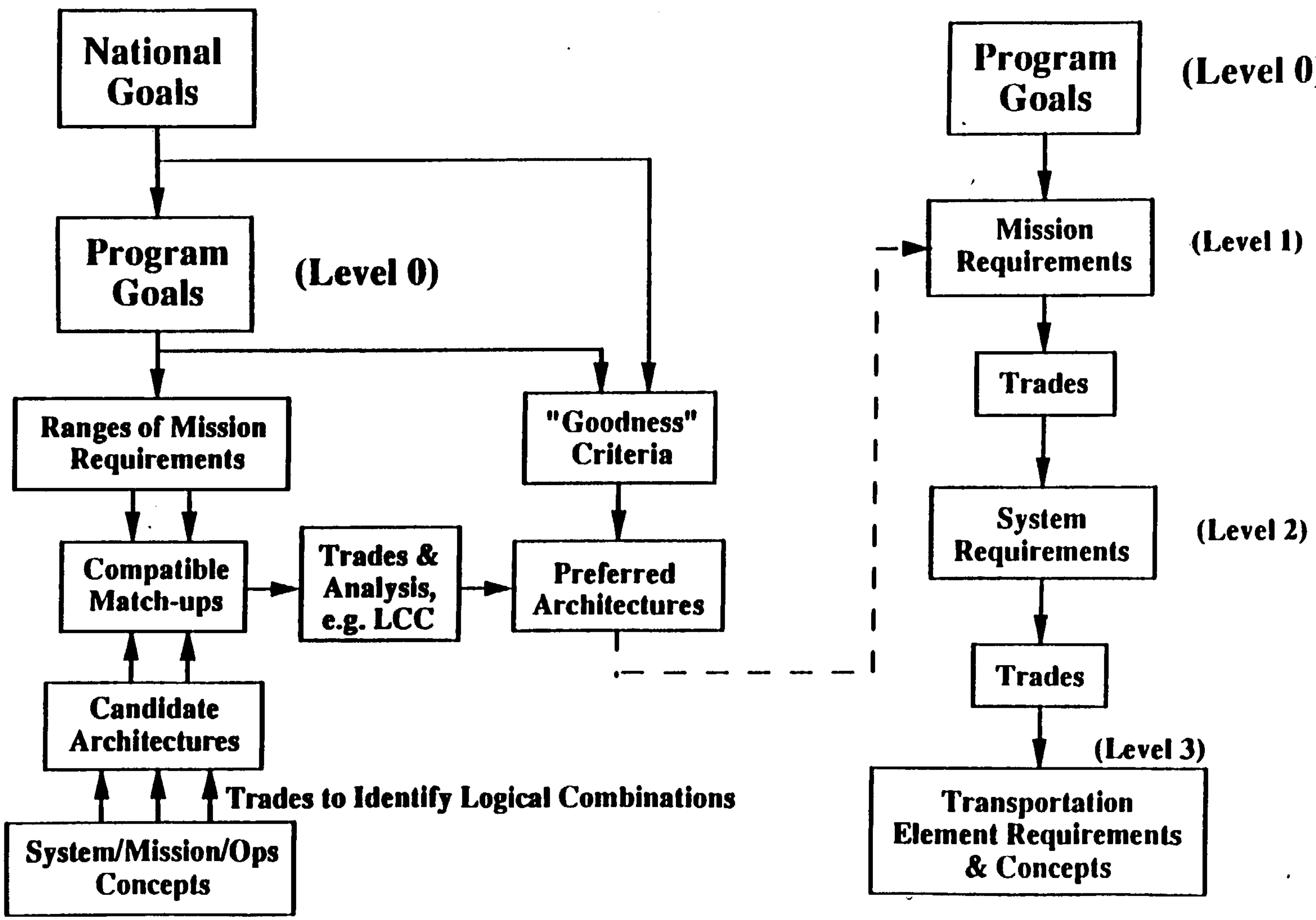
Optimization deals with infinite numbers of paths that satisfy boundary conditions. Optimization is a technique for generating only optimal paths. Any path that satisfies the boundary conditions is the sought optimal path.

Nothing quite as rigorous can be done in architecture synthesis. However, by bottom up trades, assembling systems into "good" candidate architectures, and matching with ranges of program scope, we may come close. The key is knowledge we obtain on what works well what things are compatible and combine well to satisfy mission requirements.

The last step is to conduct trades and analyses such as life cycle cost to identify preferred architectures, apply criteria derived from national goals program goals, to select among preferred architectures.

The dotted line indicates that one could then enter the traditional analysis flow with preferred architectures and their associated requirements and mission profiles, to further refine systems through systems engineering.

Architecture Synthesis vs. Mission/System Analysis



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Architecture Trade Flow

The facing page shows the low level system mission and operations trades that have been conducted or are being conducted for our seven architectures to represent the range of possible architectures for the SEI mission. Most of the trade areas have been presented in this briefing or have been presented in earlier briefings. The knowledge base in this area is fairly complete except that only very preliminary analyses have been done for the cryogenic direct mode and for cycler orbits. When these two options are completed we will be ready to finish up the architecture analysis.

Architecture Trade Flow

Cryo/Aero-braking	Nuclear Electric (NEP)	Solar Electric (SEP)	Nuclear Thermal Rocket (NTR)	L2/Lunar Oxygen	Cryo/Aero braking Direct	Cycler Orbits

• Mission design
• Reuse
• Aerobrake
  • shape
  • heating
  • GN&C
  • structures
  • assembly
• All-propulsive conj. option
• Modularity & commonality
	• Mission design
	  • trip time
	  • gravity assist
	  • node location
	• Power cycle
	• Power level
	• Specific power
	• Redundancy mgmt.
		• Mission design
		  • trip time
		  • gravity assist
		  • node location
		• Solar cell type
		• Power level
		• Specific power
		• Assembly/deployment of large space structure
			• Mission design
			• Isp and T/W sensitivity
			• Reuse
			  • tanks
			  • engines
			  • core stage
				• All-propulsive conj. option
				• Lunar oxygen benefits
				• Integration of lunar & Mars ops.
				• Advanced propulsion for LEO-L2 operations
					• Performance vs. separate MTV/MEV
					• Sensitivity to propellant choice
						• Mission design
						• Feasibility of high Mars encounter velocities
						• Design of "taxis"
						• Operational integration

For all: Overall configuration; key subsystems performance; integration compatibility; operations analyses

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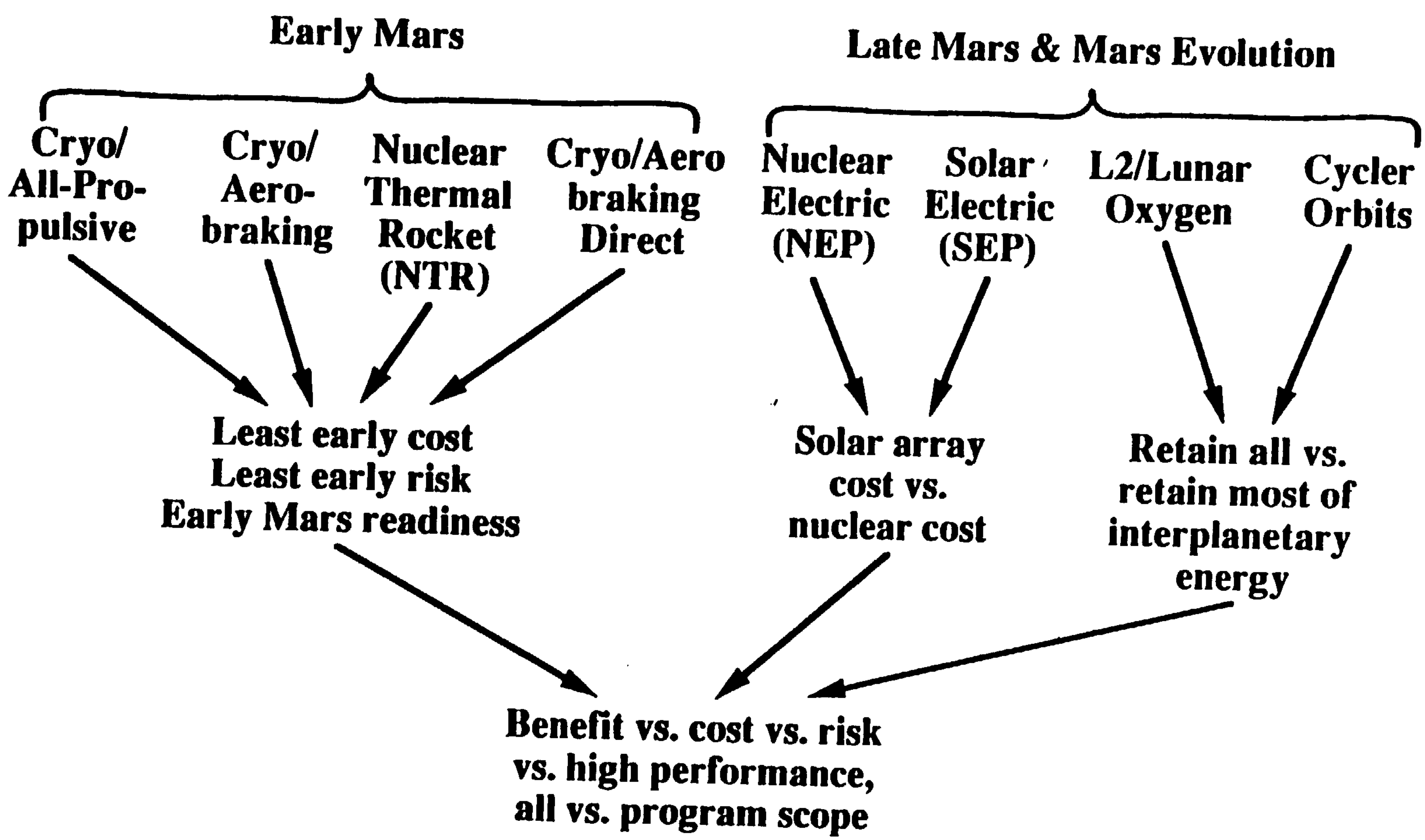
Architecture Evaluation Approach



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Mars Summary

- More than 20 beneficial modes identified.

- Early Mars: Cryo all-propulsive (CAP), ECCV*, conjunction;
  NTR all-propulsive, conjunction or opposition;
  Cryo aerobraking opposition, ECCV;
  (possibly) Direct with Mars oxygen.

- High performance, late Mars or evolution:
  SEP or NEP;
  ISRU, moon or Mars or both;
  Combintations.

- Efficiency range 10:1 measured as RMLEO (resupply mass LEO).

- Reusable MEV/Mars propellant has significant leverage for high-performance options.

* Earth Crew Capture Vehicle, an Apollo-like capsule used for Earth entry and landing or aerocapture to LEO. The rest of the vehicle is expended.

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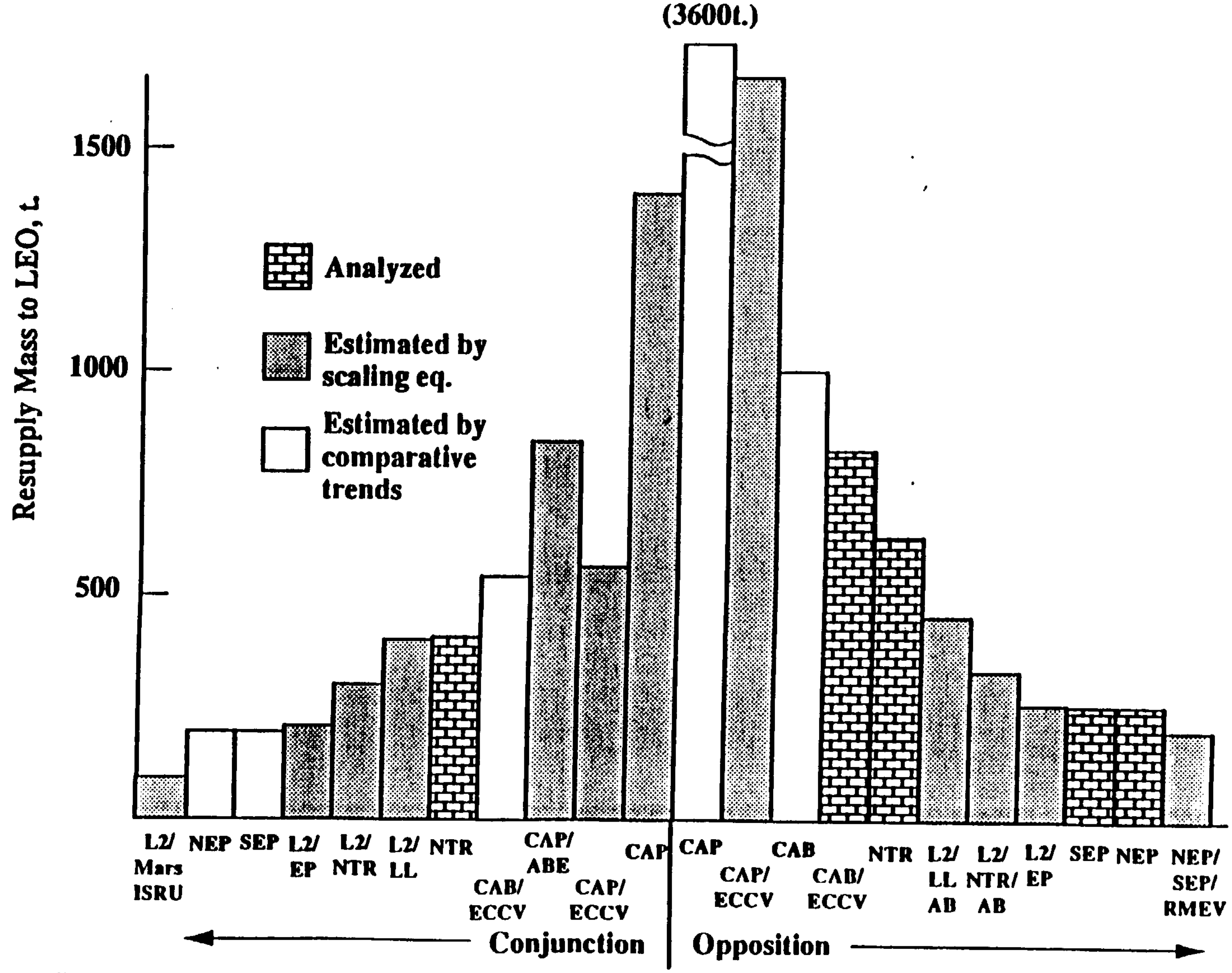
Comparison of Propulsion Options



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Conjunction vs. Opposition Mars Profiles

Opposition Advantages

- Shorter overall trip time, by at least a year.
- Transfer vehicle usually returns in time to be reused on next opportunity.
- Enables crew rotation/resupply mode with synodic period stay time.

Conjunction Advantages

- Lower energy; significantly less RMLEO unless very high Isp available.
- Venus swingby complexity not necessary.
- Long stay times at Mars.
- Shorter transfer times.
- Elliptic parking orbits can be optimized.

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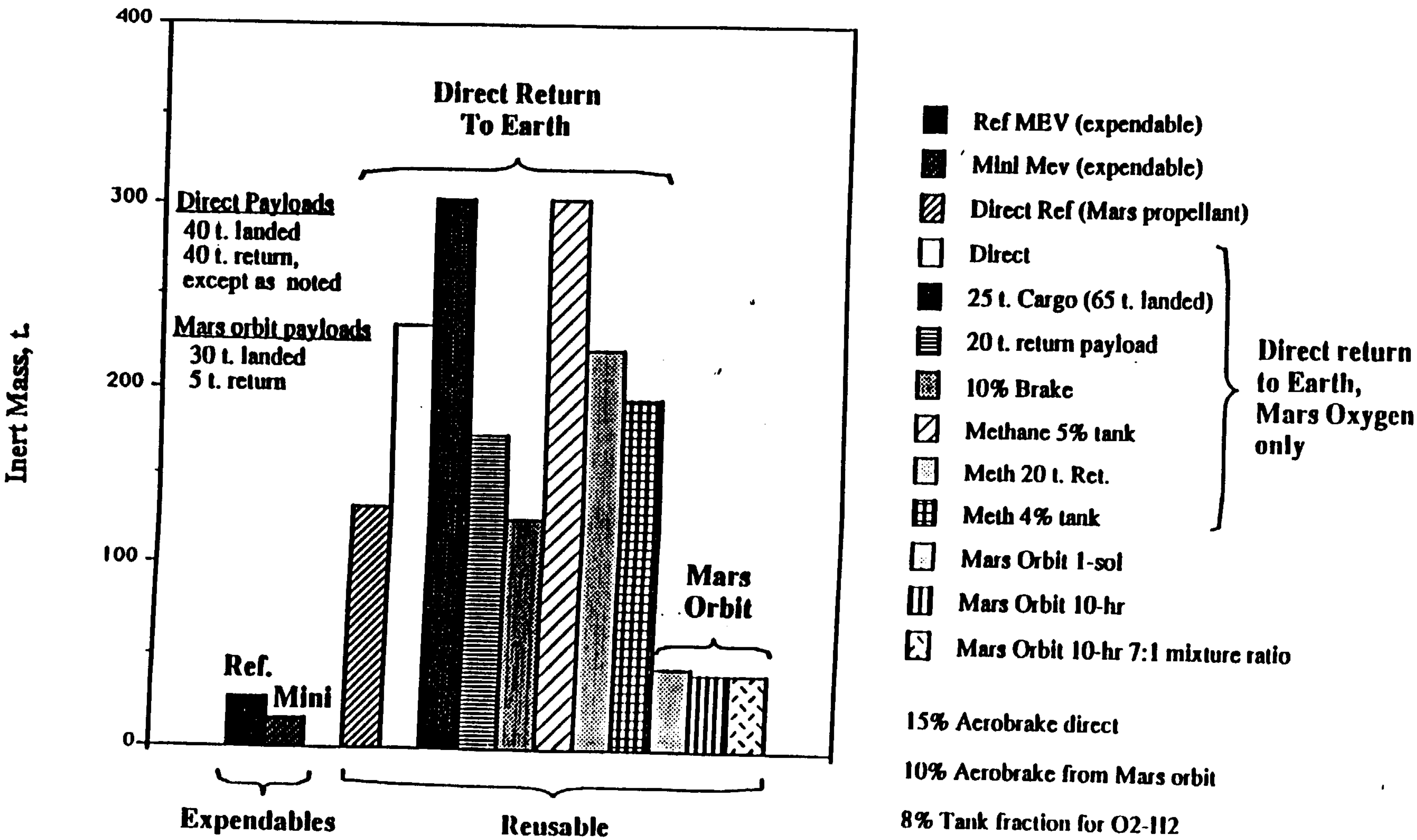
Reusable MEV Sensitivities



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II. Requirements, Guidelines and Assumptions

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Reference and Alternate Missions

Note: Contains material formerly in Mission Analysis

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NTR- Mission Analysis

The trajectories used in the NTR mission profiles are the same ones as used by the Cryo/ Aerobrake vehicle, with the exception of additional"fast- trip" time conjunction mission profiles which would require the Cryo/Aerobrake IMLEO to an unreasonable level (≥ 900 t) These "fast- trips" are in process of being generated and examined. These results will be given in the final version of this document.

The first chart and its accompanying text show an early version of the mission profile for a 2016 mission to Mars, which was the NASA derived baseline mission . The next two charts and text show the current mission profile, which is the Boeing generated profile in the same time frame as the NASA (Level II reference) profile. The changes between the two were addition of a third departure burn from Earth orbit to lower gravity losses, and a lower plane change delta V requirement with a slight change of departure and arrival dates at Earth and Mars. In addition trajectories for both conjunction and opposition missions in dates ranging from 2010 to 2025 were found and analyzed. They and their consequences, in terms of launch criteria from Earth, and arrival and orbit conditions at Mars, are given in the next three charts.

The next set of charts considers the trajectory effects of a low-pressure NTR, which has been considered in some of the architectures as a possible advanced vehicle. It has been hypothesized that at low pressures, hydrogen recombination from monatomic to diatomic will release energy in the nozzle and boost specific impulse significantly. The low operating pressure, however, forces low thrust, which in turn increases finite-burn gravity losses. The gravity loss for a three-burn departure at 1250s Isp is calculated at 311 m/s.

Another chart shows the altitude vs time relationship for the three burn departure, with the radiation belts indicated to show the duration the vehicle stays in the various radiation concentrations of the belts during the Earth escape maneuver.

The mission velocity required varies significantly with the mission start year (reference the previous trajectory information), since Mars has an orbit with high eccentricity compared to other planets. The propellant required varies in the NTR case over about a factor of two, as shown on the next chart in this section. The final series of charts gives parametric start mass curves as a function of the delta V's of the various propulsive burns.

NTR MISSION PROFILE

Once assembled in LEO at Space Station Freedom the 2016 NTR-powered Mars vehicle departs Earth utilizing a 2 or 3 burn TMI (Trans Mars injection) departure with its single 75k lbf thrust NERVA derivative engine and thus begins its 434 day journey with includes an inbound unpowered Venus swingby. The vehicle payload consists of a 73 t MEV and a 4 man 34 t MTV crew hab module. A single Earth departure burn would incur sizable g-losses due to the vehicles low overall T/W ratio (vehicle T/W=0.04). Splitting the departure burn into 2 or 3 phases and firing each time near the orbit periapsis point is to be used to decrease these g-losses losses to an acceptable level.

After TMI, the empty Earth departure propellant tanks are jettisoned as a means of lightening the vehicle for subsequent burns. Orbit capture at Mars is done all propulsively and as before, propellant tanks are jettisoned. The crew enters the MEV and descends to the surface using a combination of aerobraking and engine thrust for deceleration. After a 30 day surface stay the crew returns to the transfer vehicle via the MEV ascent stage. Once onboard, the transfer vehicle does a single TEI burn and begins the inbound journey with only the MTV crew module as payload.

After the Venus swingby andan inbound midcourse correction burn the vehicle will return to Earth in one of two ways, shown on the sketch as return option 5a or 5b. 5a is the vehicle expendable mode - the crew enters a small Apollo type ECCV (weighing 7 tons) which enters the atmosphere via heat shield aerobraking to achieve eventual splashdown in the Pacific. Option 5b represents the vehicle reuse mode - an all propulsive Earth capture burn is done to capture into a 500 km by 24 hr elliptical orbit. No ECCV is taken for this reuse mode.

ADVANCED CIVIL SPACE SYSTEMS

2016 NTR Vehicle Mission Profile

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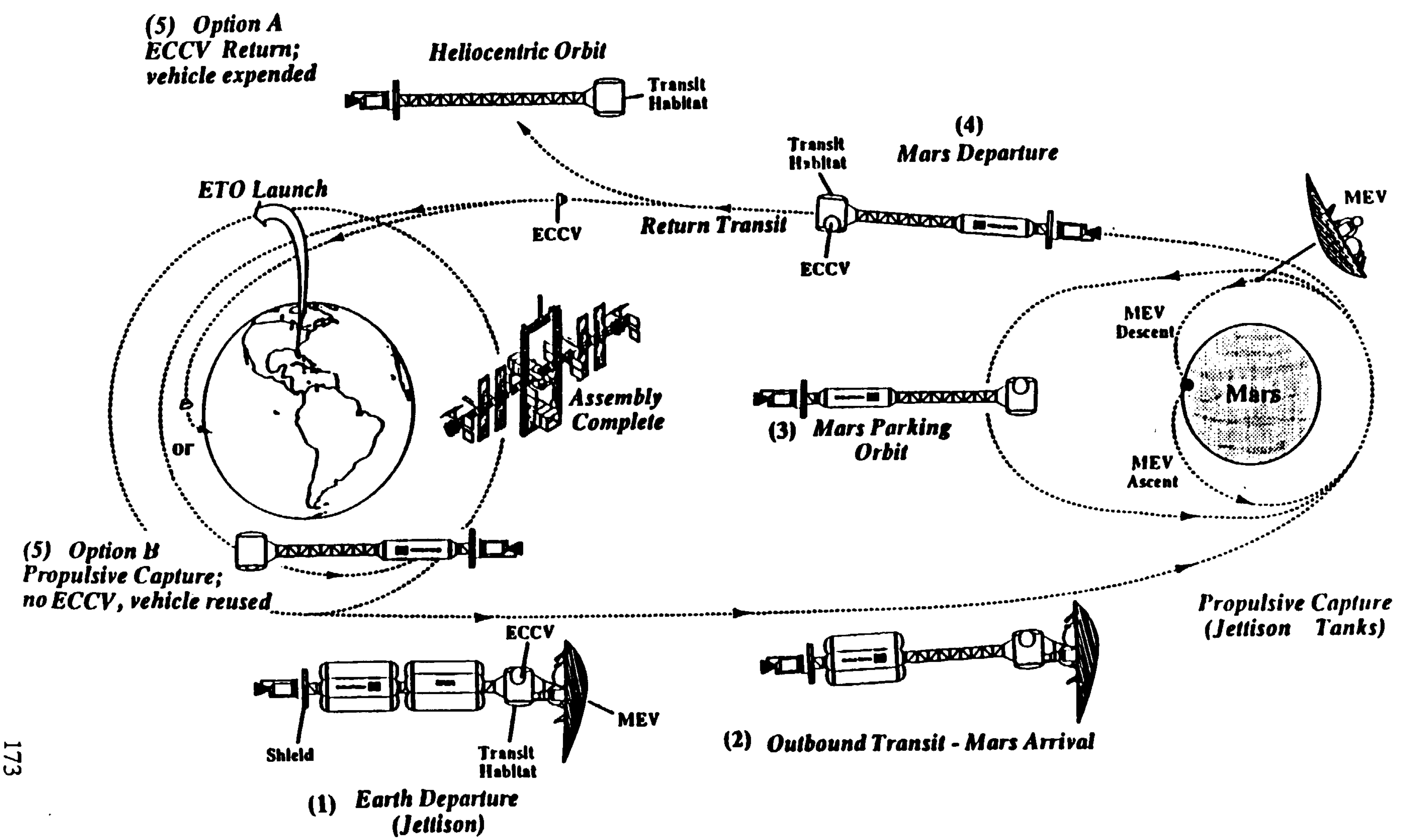


NTR Mission Profile Schematic

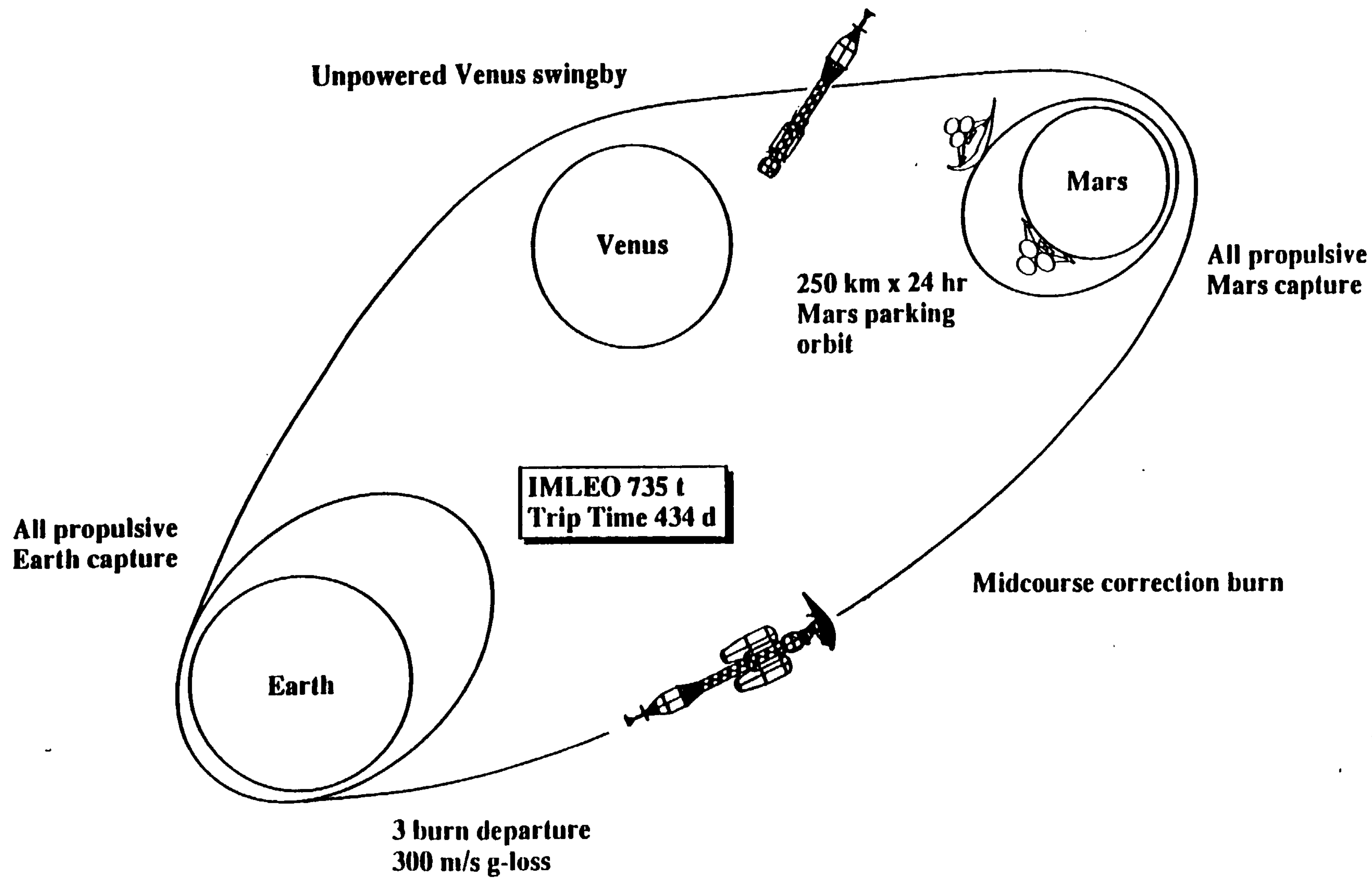
The nuclear thermal rocket mission profile is relatively simple. The NTR departs low Earth orbit using a 3-burn departure to minimize the loss. The 3-burn departure takes less than one day. The NTR coasts to Mars, is captured propulsively into a 250 km by 24-hour Mars parking orbit, following which the Mars vehicle lands crew on Mars for a surface mission. Some of these mission profiles carry multiple Mars excursion vehicles so that more than one landing is possible on one mission.

The transportation profile shown on the facing page is for an opposition mission with Venus swing-by either on the Earth- Mars trajectory or Mars-Earth trajectory. The NTR is also applicable to a conjunction mission profile with the trip times of 900 to 1000 days with stay times at Mars approximately a year.

In either case, the NTR returns to Earth by all-propulsive Earth capture to a highly eccentric orbit. The crew returns by the ECCV or by pickup by an LTV. At a later date the NTR can be brought down to Earth orbit by being refueled, or can be serviced in high orbit and operated from there.

This transportation mission profile is also applicable to a cryogenic all-propulsive conjunction mission. The cryogenic all-propulsive option is too massive on an opposition profile, but is in the reasonably competitive range for the conjunction profile.

NTR Mission Profile Schematic



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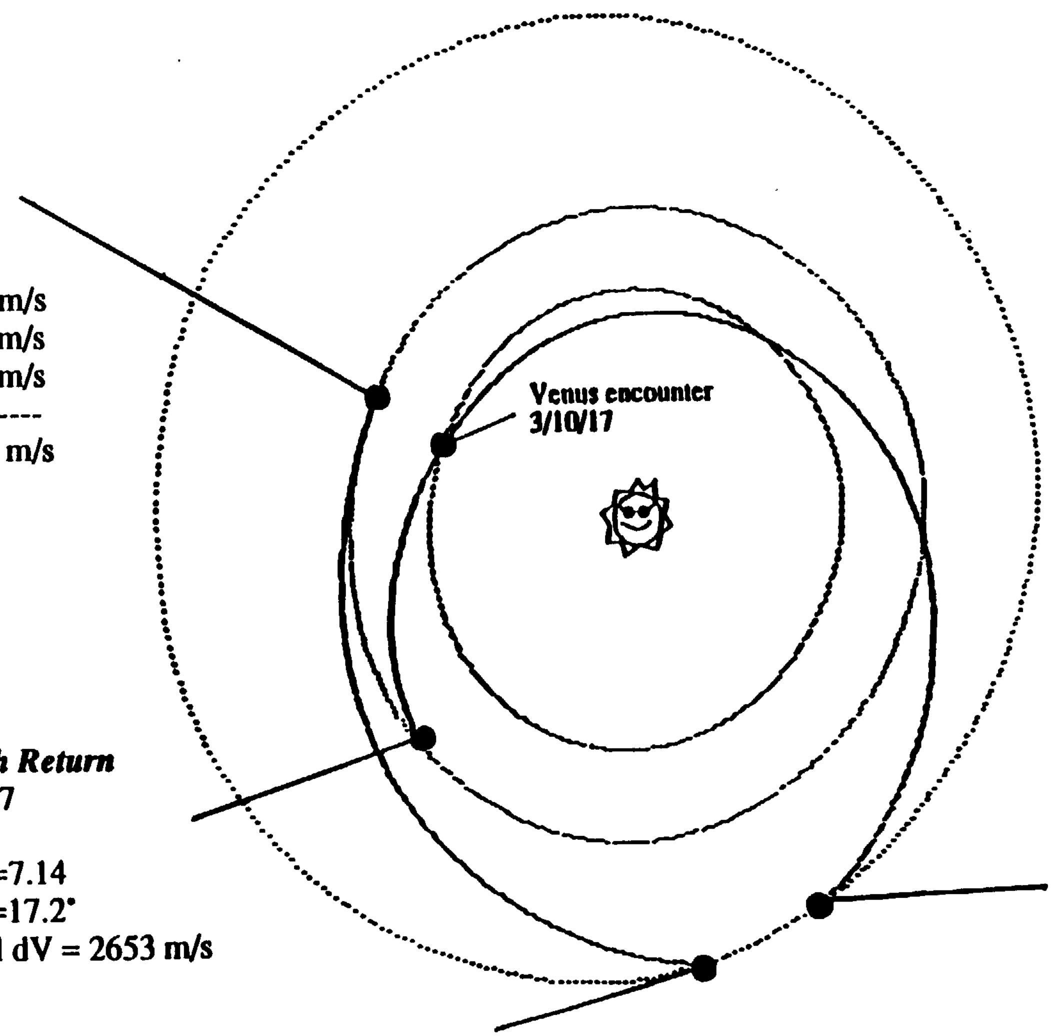
2016 NTR FLIGHT TRAJECTORY

The flight trajectory shown below was utilized as the trajectory from which all the NTR vehicle propellant loadings were based. The mission delta V's are listed for the TMI, Mars arrival, TEI and Earth arrival burns. Not listed are the outbound and inbound midcourse correction delta V's which are 120 and 90 m/s respectively. This 434 day trajectory is characterized by a 30 day Mars stay time and the inbound Venus swingby assist and was considered as a near optimum case for the 2015-2016 NTR vehicle design case study.

2016 NTR Reference Trajectory

Boeing #2 Modified/434 day/Venus Swingby



Mars Trajectory Data

STCAEM/pab/10/19/90

Opportunity (optimized runs)		Earth Dep C3	Mars Arr. Vhp	Mars Departure C3	Mars Departure ΔV	Earth Arrival Vhp	Earth Arrival C3	Mars Orbit Inc. (Deg)	Earth launch DLA	Mars Arr. LVI
2010 Opposition	12/01/10 -10/26/11, 11/25/11- 8/31/12	28.69	4.93	16.1	2.32	6.99	48.9	30	4.19	-28.63
2010 Conjunction	10/26/9 -10/31/10, 8/27/11 - 7/15/12	11.14	3.26	7.0	2.73	3.80	14.4	30	32.82	9.48
2013 Opposition	11/22/13 - 8/6/14, 10/5/14 - 8/14/15	13.08	4.10	37.7	3.54	4.53	20.5	30	20.42	25.32
2013 Conjunction	12/3/13 - 9/23/14, 9/28/15 - 9/6/16	9.58	3.15	6.9	2.61	4.47	20.0	45	23.73	31.13
2015 Level II Reference	5/23/15 - 4/22/16, 5/22/16 - 12/8/16	20.19	7.02	30.0	4.70	8.77	77.91	30	55.92	14.51
2015 Level II Alternate	10/15/15 - 7/16/16, 8/15/16 - 5/17/17	48.38	4.79	33.3	3.58	3.94	15.5	30	0.18	15.3
2015 Conjunction	12/24/13 - 11/17/14, 12/14/15 -10/8/16	8.89	4.22	5.4	2.37	5.52	30.5	35	18.45	32.93
2015 L II Ref. + 50 day	5/23/15 - 4/22/16, 5/22/16 - 1/27/17	14.19	6.93	18.0	2.37	9.47	89.7	30	55.92	15.22
2016 Boeing Nominal	2/25/16 - 7/31/16, 8/31/16 - 5/5/17	10.34	6.82	39.7	4.37	7.14	51.0	30	-35.94	-1.69
2018 Opposition	3/27/17 - 3/10/18, 4/24/18 - 12/18/18	19.71	5.96	10.9	2.58	5.04	25.4	30	17.97	22.3
2018 Conjunction	5/12/18 -11/28/18, 5/31/20 - 11/27/20	7.86	2.97	7.9	3.41	4.02	16.2	45	-37.94	-7.81
2020 Opposition	6/4/20 - 12/11/20, 1/10/21 - 1/28/22	24.40	3.89	27.8	3.77	4.28	18.3	20	15.1	-9.69
2020 Conjunction	7/20/20 - 1/16/21, 8/09/22 - 1/16/23	13. 40	3.13	18.8	3.92	6.67	44.5	30	18.65	-3.40
2022 Opposition	11/11/21 - 9/17/22, 10/17/22 - 6/5/23	16.31	5.31	43.2	4.46	6.00	36.0	20.	-60.02	-4.04
2023 Conjunction	9/8/22 - 4/16/23, 7/9/24 - 5/5/25	19.03	3.18	12.3	2.66	2.86	8.81	30	50.73	23.71
2024 Opposition	9/20/23 - 7/4/24, 8/4/24 - 5/30/25	27.91	6.46	9.0	1.61	3.08	9.49	30	-19.58	-6.53
2025 Conjunction	10/17/24 - 6/24/25, 8/11/26 - 5/5/27	19.68	3.00	8.3	2.60	2.60	6.76	35	55.88	34.75

Level II Reference : ΔV Earth Departure = 4281 m/sec ΔV Earth Arrival = 6278 m/sec (at LEO)
Data From MASE ΔV Mars Arrival = 3949 m/sec ΔV Mars Departure = 3400 m/sec

ADVANCED CIVIL SPACE SYSTEMS

Trajectory Information for Mission Opportunities

BOEING

STCAEM/pth/19Mar90

Opportunity	C3 Earth Departure	Vhp Mars Arrival	C3 Mars Departure	Vhp Earth Arrival	Periapsis Altitude (km)	Apoapsis Radius (km)	Periapsis Radius (km)	Eccentricity	Semi-major Axis (km)
2010 Opposition	28.68	4.93	16.07	6.99	250	37188.13	3647	0.82	20415.57
2010 Conjunction	11.14	3.26	7.04	3.72					
2013 Opposition	13.07	4.10	37.68	4.53					
2013 Conjunction	9.58	3.15	6.88	4.47					
2015 Level II Reference	20.19	7.01	29.97	8.76					
2015 Level II Alternate	48.36	4.79	33.28	3.94					
2015 Conjunction	8.89	4.22	5.42	5.52	500	36932.86	3897	0.81	
2015 L II Ref. + 50 day	14.21	6.93	17.97	9.47	250	37182.86	3647	0.82	
2016 Boeing Nominal	10.34	6.82	39.72	7.14				0.82	
2018 Opposition	19.71	5.96	10.93	5.04		17243.37		0.65	10443.19
2018 Conjunction	7.86	2.97	12.28	4.02		37188.13		0.82	20415.57
2020 Opposition	24.39	3.89	27.83	4.28		36521.48		0.81	
2020 Conjunction	13.40	3.89	20.03	6.67		37188.13		0.82	
2022 Opposition	16.31	5.31	43.16	6.00		36521.48		0.81	
2023 Conjunction	19.03	3.18	9.31	2.86		37188.13		0.82	
2024 Opposition	27.91	6.46	9.02	3.08		37188.13		0.82	
2025 Conjunction	19.68	3.00	7.81	4.79	250	37188.13	3647	0.82	20415.57

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Orbit Insertion Lighting Angle, Latitude, and Approach Turning Angle for 2010 to 2025 Mars Mission Opportunities

Opportunity	Periapsis Lighting Angle (°)	Periapsis Latitude (°)	Approach Turning Angle (°)
2010 Opposition	54.19	1.21 N	70.96
2010 Conjunction	42.20	42.51S	58.30
2013 Opposition	21.94	24.33 S	65.70
2013 Conjunction	55.01	16.40 S	57.22
2015 Level II Reference	22.57	28.22 S	78.88
2015 Level II Alternate	35.45	29.97 S	70.19
2015 Conjunction	67.75	22.12 S	67.58
2015 L II Ref. + 50 day	23.27	27.96 S	78.67
2016 Boeing Nominal	11.15	28.88 S	78.37
2018 Opposition	36.53	24.21 S	75.60
2018 Conjunction	50.52	47.65 S	55.16
2020 Opposition	13.50	15.97 S	63.94
2020 Conjunction	10.67	22.59 S	57.01
2022 Opposition	66.41	26.90S	72.90
2023 Conjunction	10.61	1.99 N	57.50
2024 Opposition	68.29	26.75 S	77.31
2025 Conjunction	15.32	22.06 S	55.55

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NTR 900 Isp Staged Tanks and Engines, Mode 6

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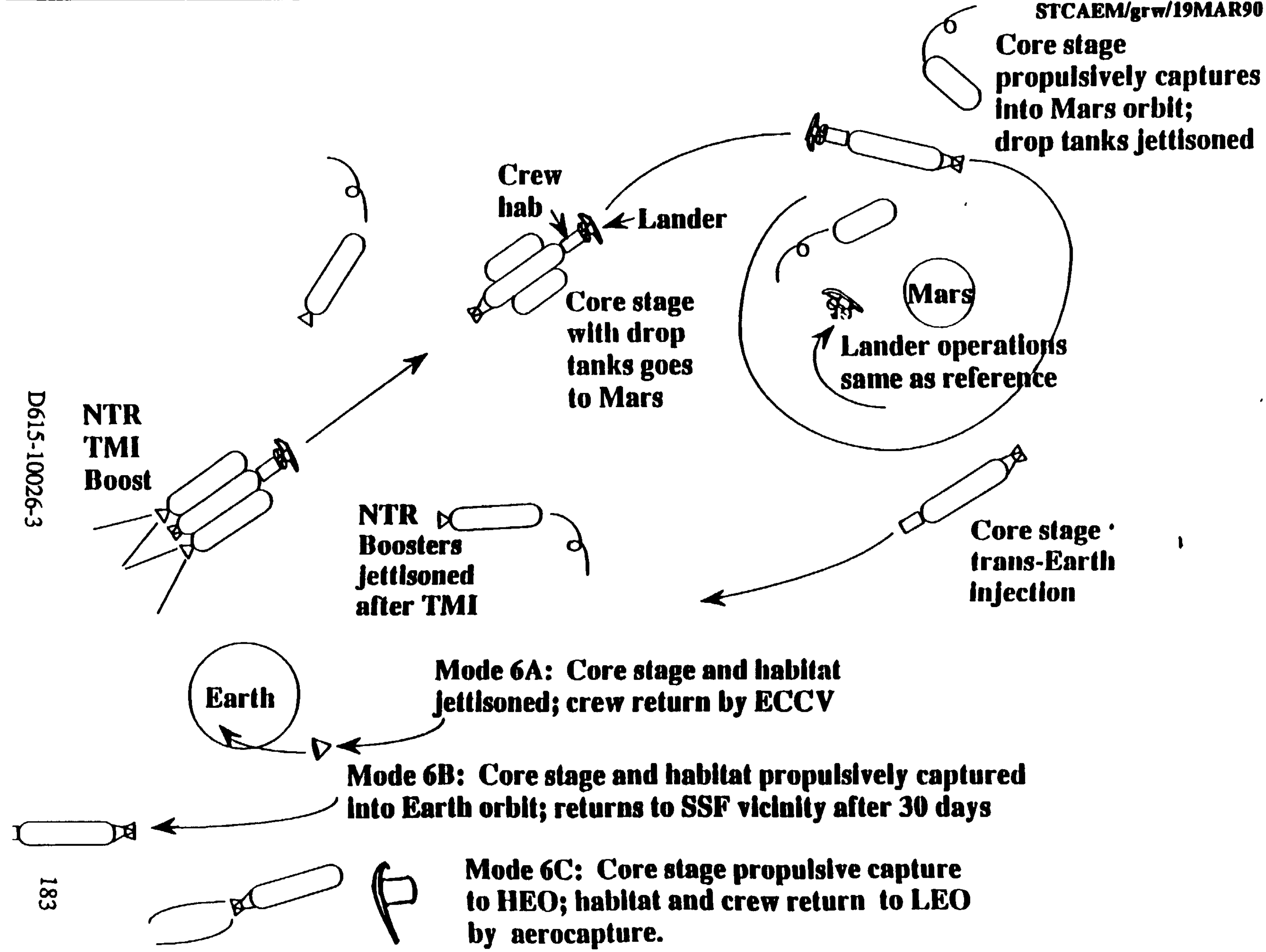

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Multiperiapsis Earth Departure Burns For Moderate to Low Vehicle T/W

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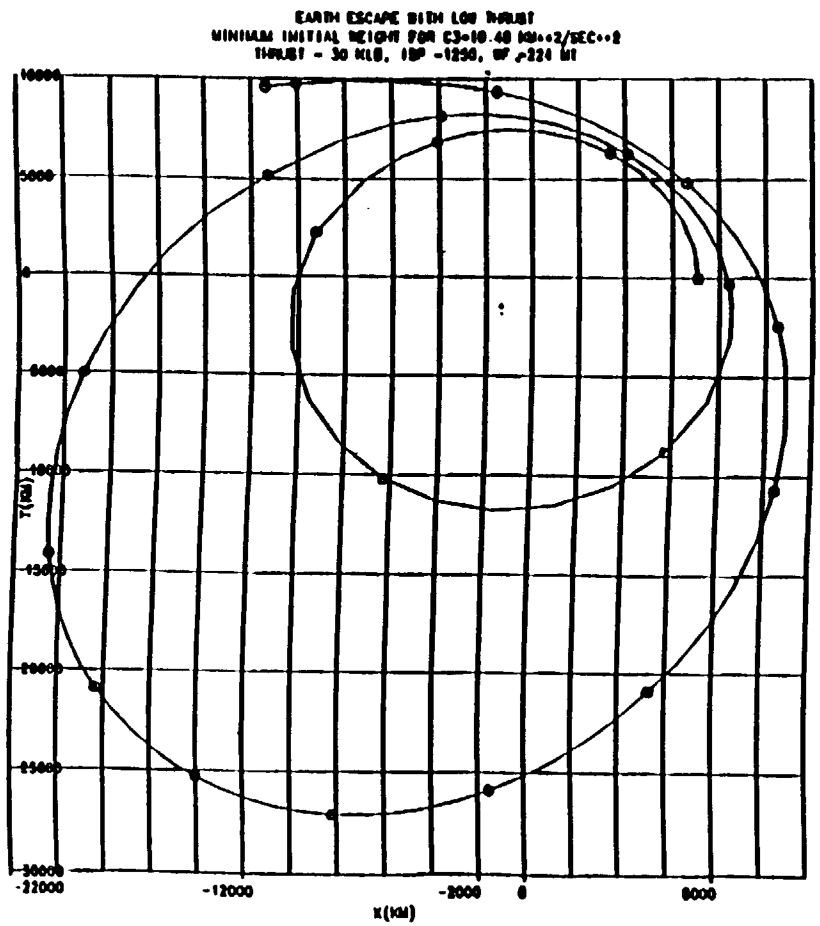
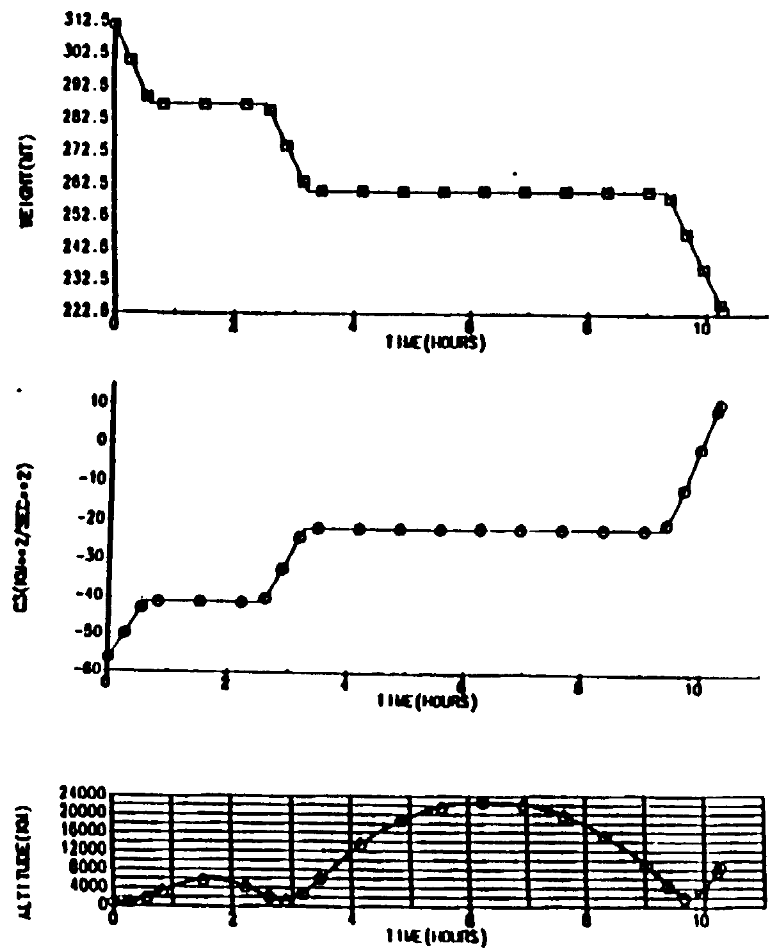
Representative case of low thrust 1250 Isp NTR sytem, 3 10k lbf engines Vehicle T/W = 0.04

3 burn Earth departure with 107 t MEV/MTV payload and 224 t Earth dep cruise mass; G-loss calc = 311 m/sec





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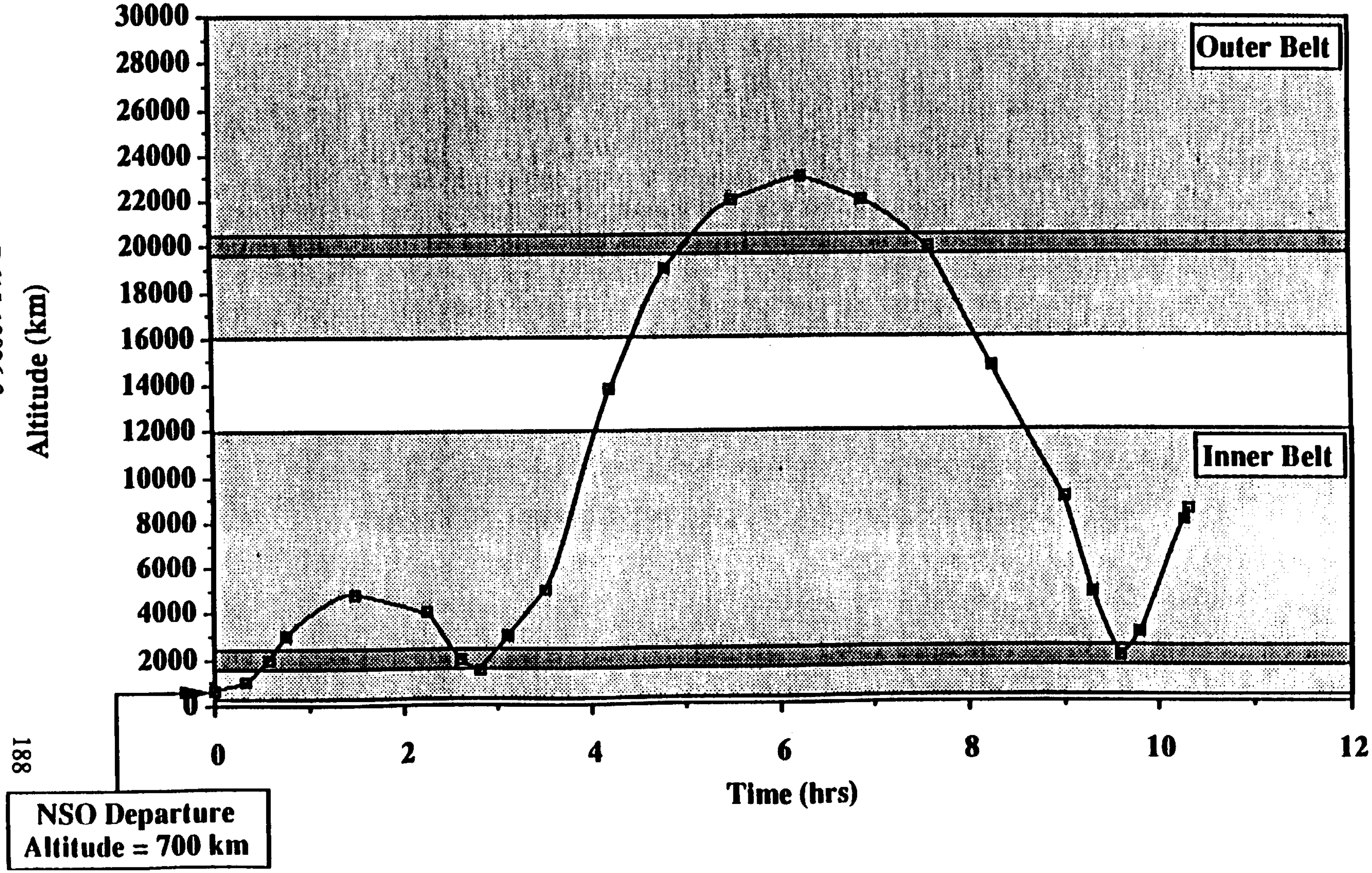
Performance Parametrics

Note: Contains material formerly in Mission Analysis

Low Thrust NTR - 3 Burn TMI - Altitude vs. Time

- Altitude parameters of inner and outer radiation belts will vary with solar activity and changes in latitude



ADVANCED CIVIL SPACE SYSTEMS

Nuclear Systems/Nuclear Solid Core Reactor Unique Issues

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- Reactor radiation shielding
- Post burn reactor 'cooldown' requirement
- Current single NTR engine preference precludes traditional engine out margin
  - relatively heavy and costly reactor systems
  - multiple point source for radiation undesirable
- Full up engine test program concerns more complex now than NERVA era
- Very large hydrogen propellant tanks dominate vehicle physical configuration

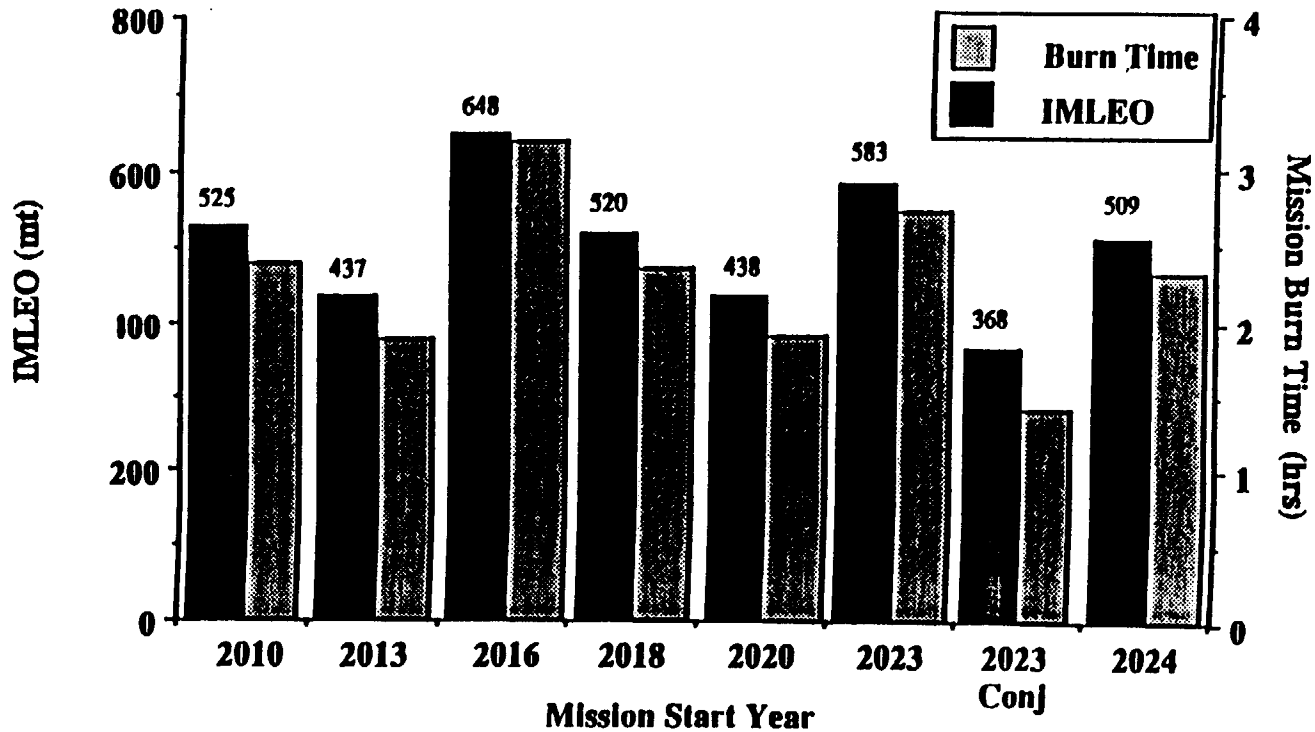
NTR vehicle weight vs opportunity year

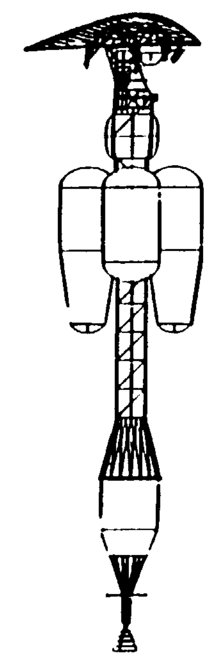
An earlier Boeing reference 2016 NTR vehicle (slightly lighter than the present 735 t reference) that utilized a NERVA derivative engine of 75000 lbf thrust, and a mass of 9684 kgs (t/w=3.5) was, for this particular trade, replaced with a Particle Bed Reactor (PBR) of the same thrust but with a t/w=10 (mass = 3401 kg). IMLEO figures for this modified craft were determined for 8 missions in the 2010 to 2024 time frame using the Boeing vehicle synthesis model. The results show that the 2016 reference mission trajectory proved to require the most propellant of all those evaluated. Engine burn time in hours is also listed on the chart with the IMLEO figure.

NTR Vehicle Weight vs. Opportunity Year

for Reactor/Engine T/W = 10

Revision 2 5/15/90





Engine Characteristics

- Eng T/W=10 Eng weight=3402 kg
- Engine Thrust = 75,000 lbf
- Engine Isp = 925 sec
- Reactor shadow shield wt = 4.5mt

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Vehicle Characteristics

- Opposition missions except 2023 conjunction
- 3 burn Earth departures: g-loss: 300 m/s
- Crew of 4, 33 t MTV, 76 t MEV includes 25mt surf p/l
- Propulsive capture at Earth into 500km 24hr elliptical orbit

synthesis model run #: marsntrmtv dat;123 130
Mac chart: NTR IMEO/burn time/yr 5/15/90

(3)

Reference NERVA NTR design delta V parametric data

Vehicle IMLEO is plotted vs Earth departure dV (m/s) and Mars capture dV (m/s) for a range of Mars departure dV's (m/s). The Boeing reference NERVA NTR vehicle configuration was used with the following vehicle characteristics and assumptions:

(1) ECCV crew return, vehicle expended

(2) MTV crew of 4 habitat module consists of the following:
- a. dry crew hab wt = 28531 kg. includes: 1802 kg rad shelter, 1530 kg external airlocks (2) 1539 kg external solar array power system
- b. consumables = 4422 kg (4 crew for 434 days) The data does not account for consumables weight variation with changes in mission duration
- c. on board 'resupply' mass = 986 kg
- d. transfer science equipment, hardware, supplies etc = 1000 kg

Total MTV crew module mass = 43939 kg

(3) LO2/LH2; 2 TMI tanks, 2 MOC tanks, single TEI tank at an approximate tank fraction of 14%
(4) N2O4/MMH storable RCS system, Isp=280 sec
(5) no artifical-g
(6) 120 m/sec outbound midcourse correction burn, 90 m/sec inbound
(7) MEV is propulsively captured at Mars with main NTR stage
(8) propellant boiloff was calculated for a 434 day mission; i.e. this data takes no account of boiloff variation with changes in mission duration

Reference NERVA NTR Design Delta V Parametric Data

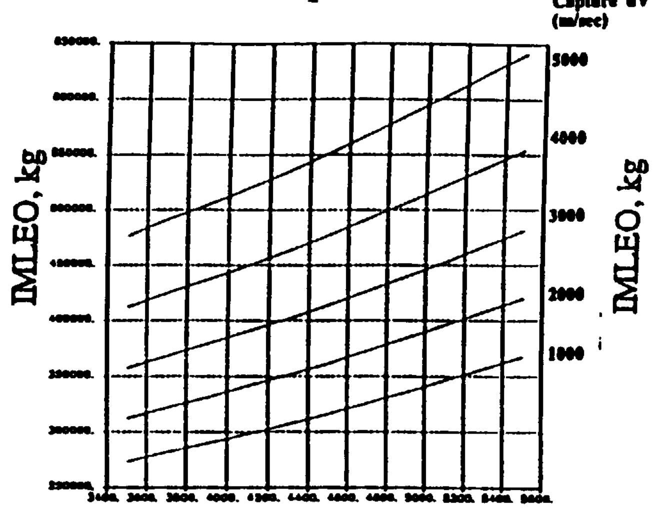
Vehi IMLEO vs E dep dV, Mars capt dV & Mars dep dV, ECCV Ret

ADVANCED CIVIL SPACE SYSTEMS — ***BOEING***

Vehicle Characteristics

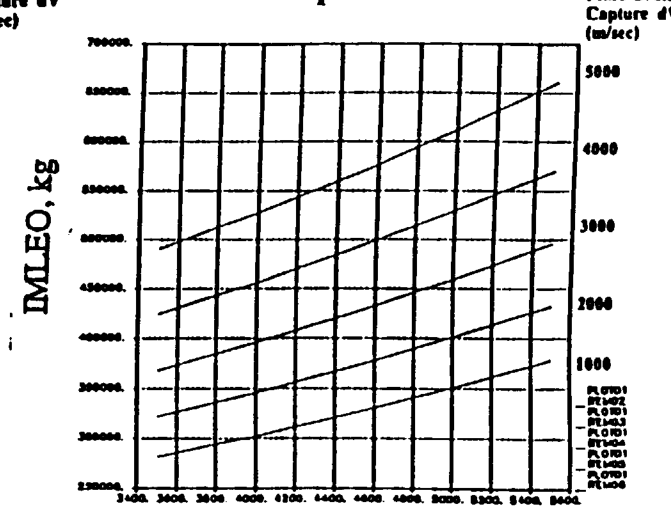
Isp=925
MEV w heat shield= 73118 kg
MTV module total = 34939 kg
NERVA engine wt = 9684 kg
Shadow shield wt = 4500 kg
Truss & tank struts=3000 kg
Five 10 m dia tanks, 14% t.f.
ECCV crew return =7000 kg
2 midcourse corr burns
Zero-g, storable RCS prop
Departs from LEO
Mars orbit: 250 km by 1 sol
consumables variation with trip time not accounted

Mars dep dV = 500



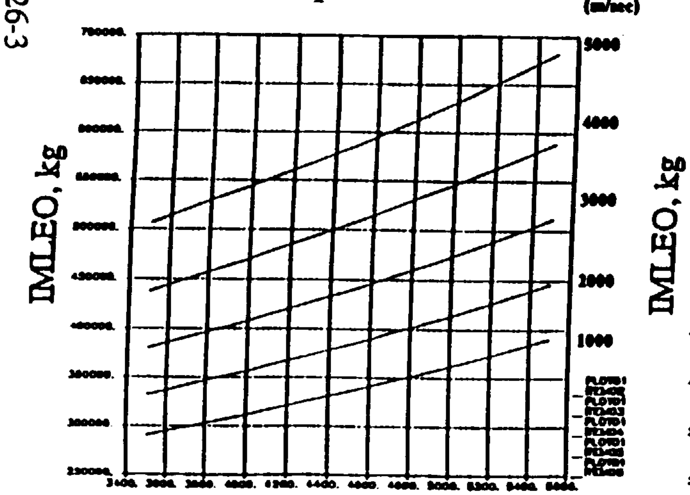
Earth dep dV (m/s)

Mars dep dV = 1000



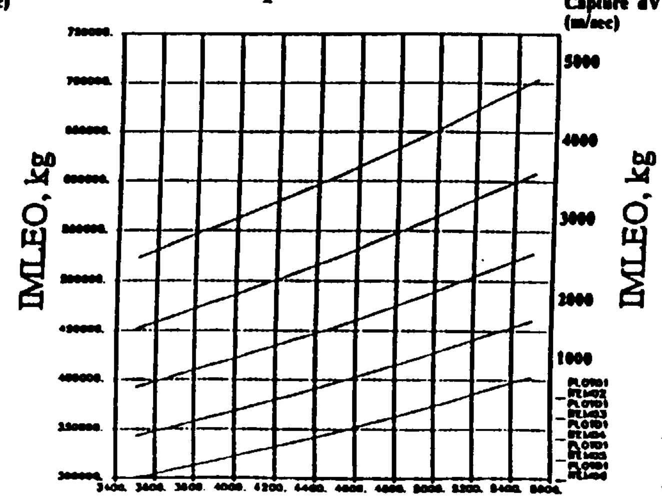
Earth dep dV (m/s)

Mars dep dV = 1500



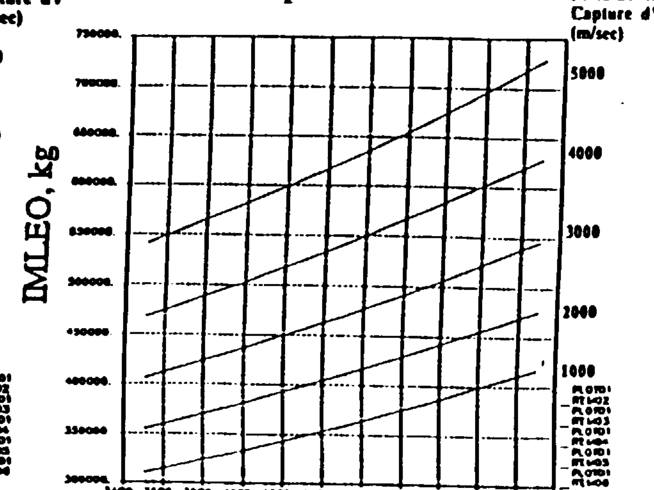
Earth dep dV (m/s)

Mars dep dV = 2000



Earth dep dV (m/s)

Mars dep dV = 2500



Earth dep dV (m/s)

Reference NERVA NTR Design Delta V Parametric Data

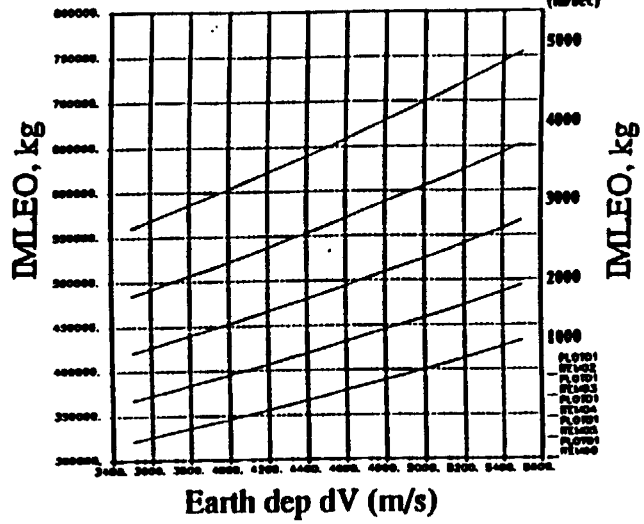
Vehi IMLEO vs E dep dV, Mars capt dV & Mars dep dV, ECCV Ret

ADVANCED CIVIL SPACE SYSTEMS ***BOEING***

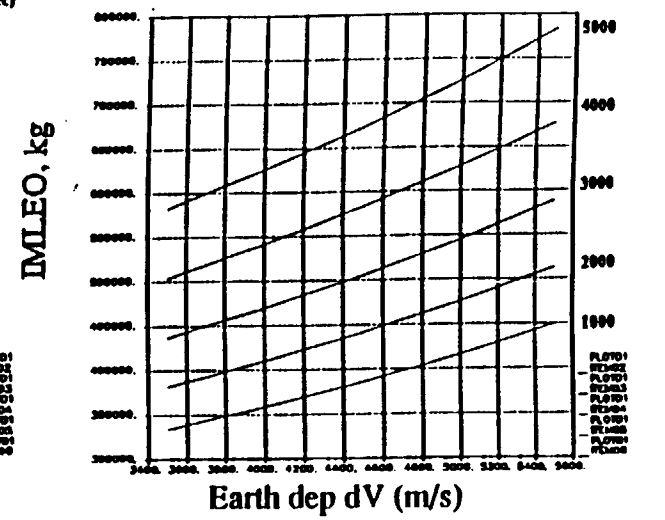
Vehicle Characteristics

Isp=925
MEV w heat shield= 73118 kg
MTV module total = 34939 kg
NERVA engine wt = 9684 kg
Shadow shield wt = 4500 kg
Truss & tank struts=3000 kg
Five 10 m dia tanks, 14% t.f.
ECCV crew return =7000 kg
2 midcourse corr burns
Zero-g, storable RCS prop
Departs from LEO
Mars orbit: 250 km by 1 sol

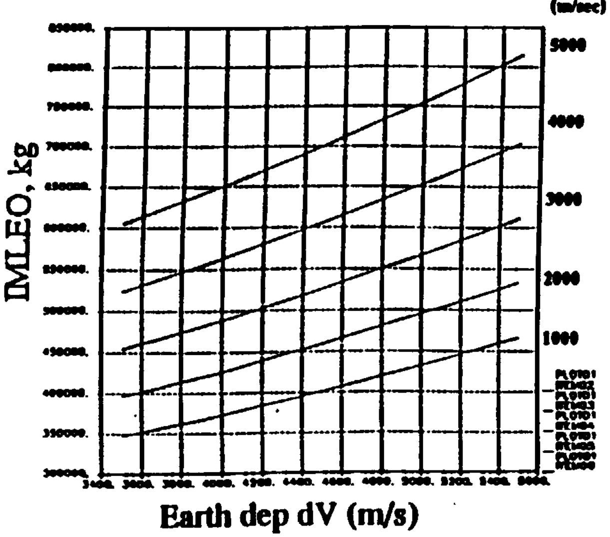
Mars dep dV = 3000



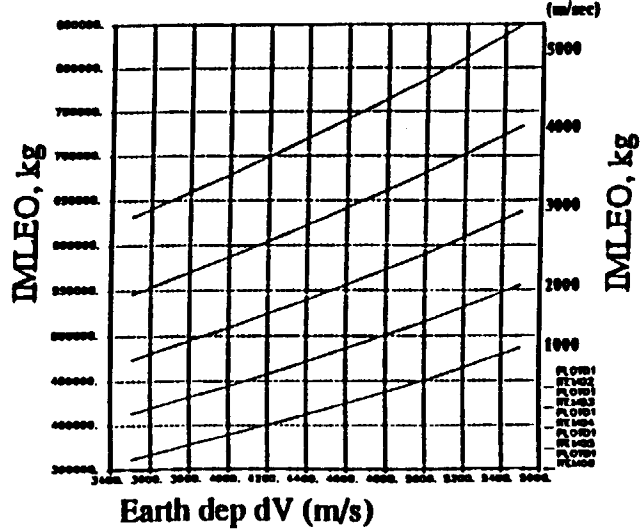
Mars dep dV = 3500



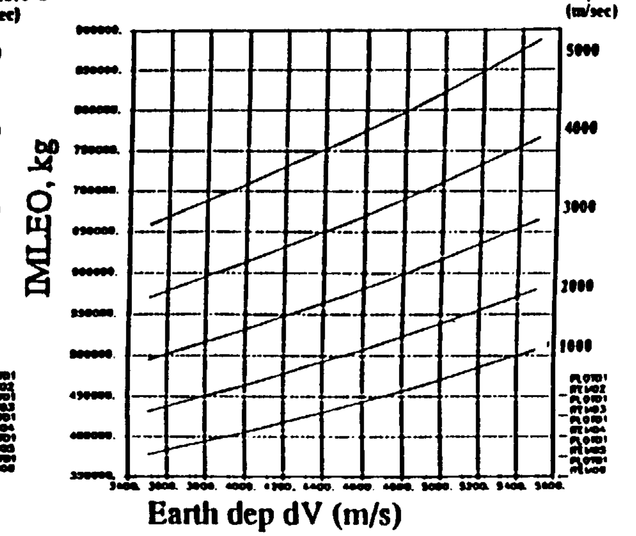
Mars dep dV = 4000



Mars dep dV = 4500



Mars dep dV = 5000



consumables variation with trip time not accounted

2018 & 2025 Non-Venus Swingby Opposition Missions

year	trip time days	Cryo/Aerobrake *Isp=475, 15% A/B*	NERVA NTR *Isp=925, eng T/W=3.5*	Advanced NTR *Isp=1050, eng T/W=20*	Reusable *Advanced NTR*
2018	350	882 t	667 t	507 t	670 t
2018	400	598 t	469 t	380 t	501 t
2018	450	589 t	416 t	342 t	423 t
2018	500	719 t	489 t	394 t	501 t
2025	350	∞ t	2,155 t	1,353 t	1,637 t
2025	400	3,804 t	1,399 t	963 t	1,080 t
2025	450	1,357 t	921 t	684 t	727 t
2025	500	1,091 t	776 t	590 t	613 t

'easy year'

'hard year'

Mac Chart/disk #8/adv propul tabular data/11-28-90

Mission Delta V's and Departure Dates-

2018 & 2025 Non-Swingby Opposition Missions

dep & arr dates: total trip time, Earth dep, Mars arr, Mars dep, Earth arr

Vhp limit at Mars capture=7 km/s, ECCV Earth entry Vhp limit=9.7 km/s
Δ = diff between arr Vhp & required arr Vhp limit

	total trip time	Earth dep	Mars arr	Mars dep	Earth arr	(1) TEI dV	Outb Deep Space burn	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV
2018 'easy year'	350	*8280	385	415	630	4.101	0	6.857	0	3.772	5.460	9.524	0	4.027
	400	8270	400	430	670	3.741	0	5.260	0	2.528	3.763	10.217	0.517	4.506
	450	8230	400	430	680	3.610	0	4.521	0	1.857	3.646	11.000	1.300	3.747
	500	8140	375	405	640	4.489	0	4.590	0	2.098	3.603	9.523	0	4.046
2025 'hard year'	350	**0390	595	625	740	10.228	0	7.020	0.020	3.984	6.223	8.050	0	3.119
	400	0365	579	609	765	7.805	1.613	7.156	0.156	4.018	4.193	4.744	0	1.428
	450	0300	540	570	750	5.187	2.761	7.229	0.229	4.079	2.030	6.020	0	2.000
	500	0275	391	421	775	4.374	2.925	6.761	0	3.693	1.946	3.749	0	1.541

(1) *g-losses not accounted for* (2) all aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo chemical propulsion to slow veh down to Vhp=7 (km/s) for aerocapture
(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo chemical propulsion to slow ECCV capsule down to Vhp=9.7 for entry

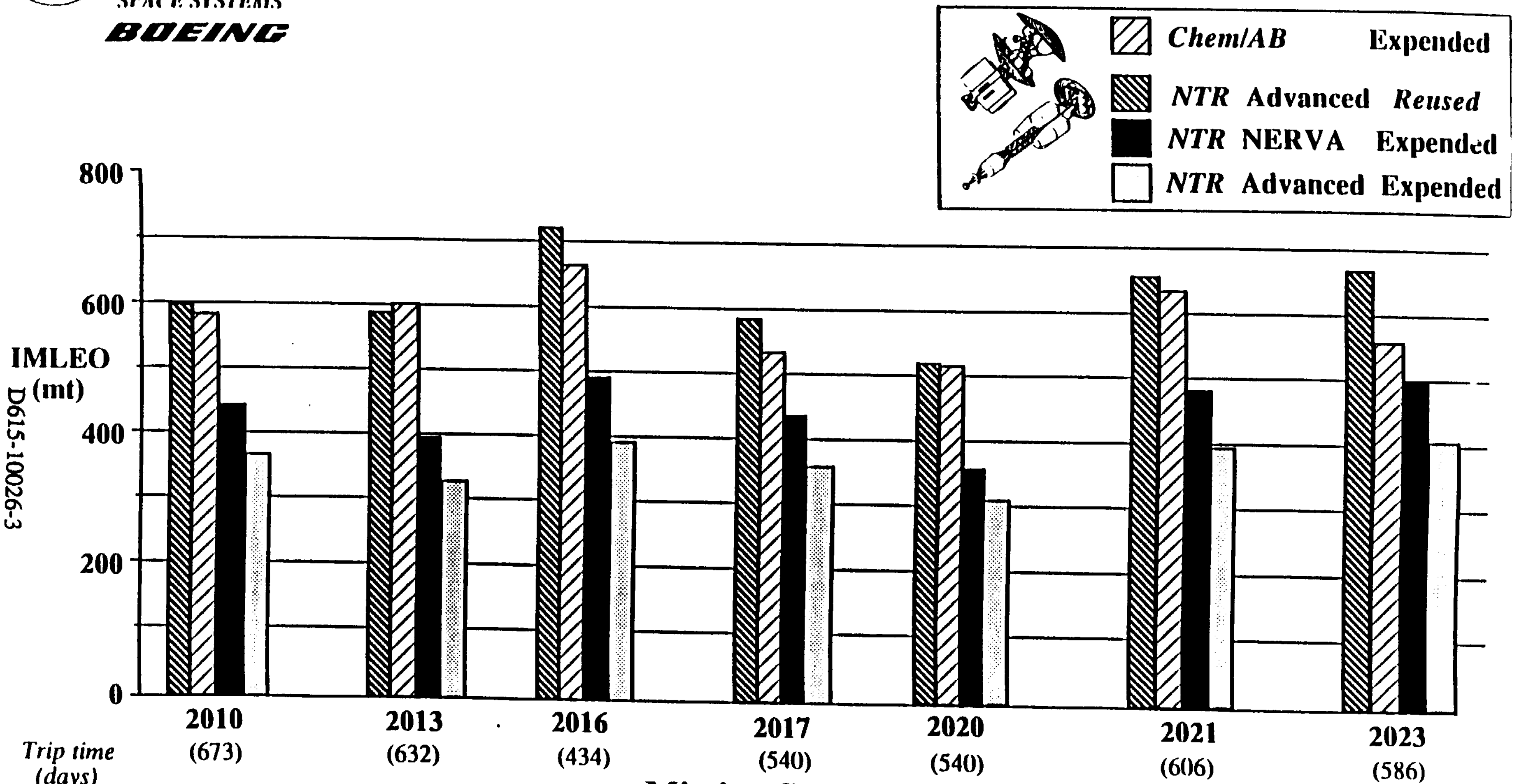
*245xxxx -, ** 246xxxx *Julian dates*

all missions 30 day Mars orbit stay time

disk #8/dVs 2018-2025 non swby

ADVANCED CIVIL SPACE SYSTEMS
BOEING

Chem/AB & NTR Vehicle IMLEO vs. Opportunity Year
Venus Swingby Opposition Missions



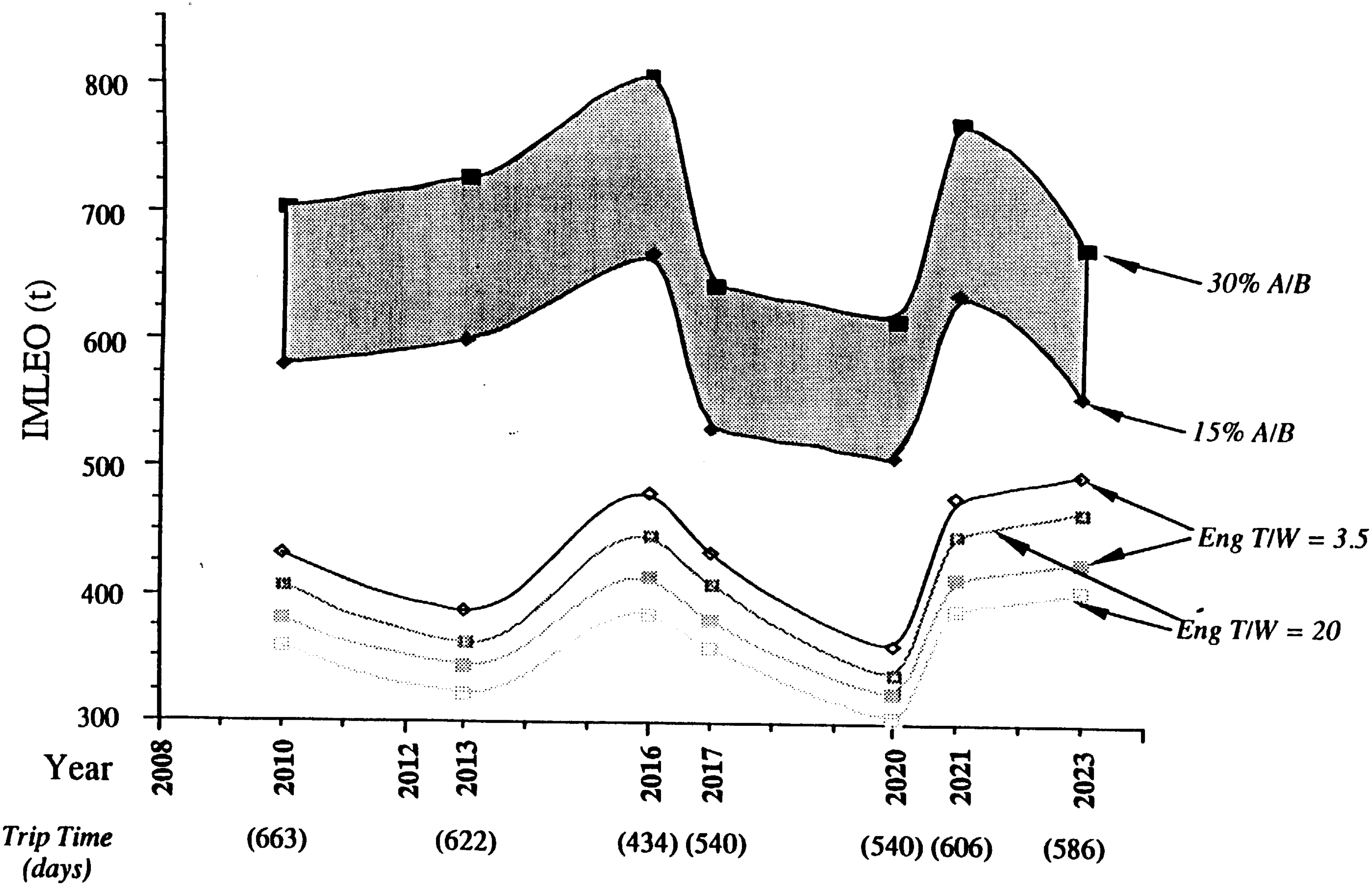
Engine & Vehicle Characteristics

- *Chem/AB: AB 15% of capt wt, eng Isp =475*
- *Adv NTR eng T/W= 20, eng wt = 1701 kg, Isp=1050 s*
- *NERVA eng T/W= 3.5, eng wt = 9684 kg, Isp= 925 s*
- *NTR engine Thrust = 75,000 lbf, Rad shield wt = 4.5 t*
- *NTR postburn cooldown prop = 3%, reserves = 2%*
- *NTR:tank fraction=14%, frame truss & t struts=2880 kg*
- *Crew of 4, 34 t MTV, 73 t MEV includes 25mt surf p/l*
- *reuse case: Propul capt at Earth: 500 km by 24 hr orbit*

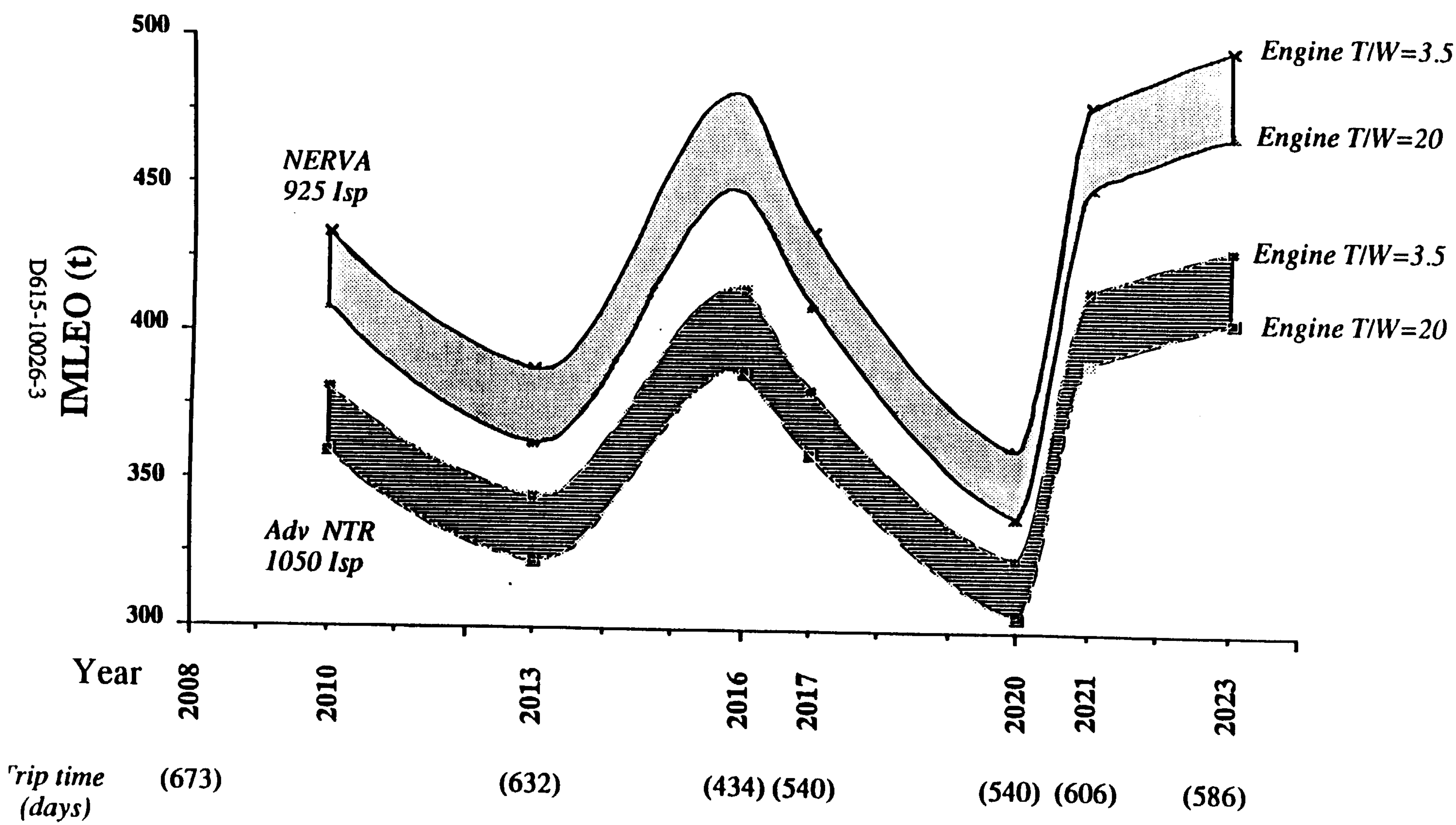
All surface stay times are 30 days

synthesis model run #: marsntrmtv.dat;345-353,369-376; marschemmtv.dat;143-151 Mac chart: Disk #5/IMLEO/vr

NTR vs Chem/AB vehicle IMLEO Comparison Opposition Missions With Swingbys



Adv NTR and NERVA NTR Vehicles

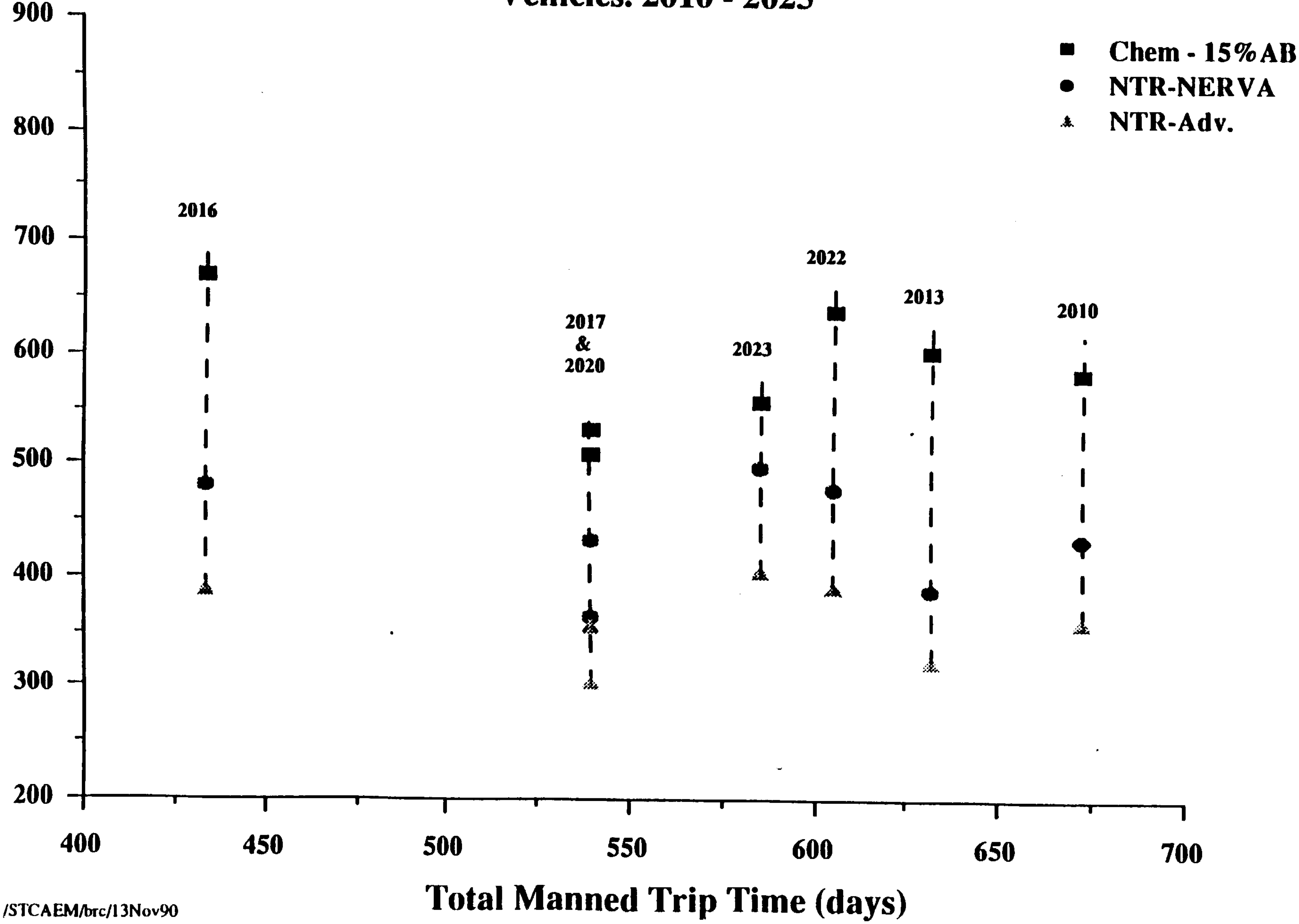


Propulsion Option swingby Opportunities for Opposition Missions

Optimum Mars trajectories with Venus swingbys are presented in the form of IMLEO vs trip time. The swingby points are given for three high trust vehicle options: Chem/AB, NERVA NTR, and advanced NTR. The swingby points are represented here as discrete points, not as a "band" . The two preceding IMLEO charts presented the swingby points as being part of a band, which is in a sense misleading since there is no continuity between two swingbys or a swingby and non swingby reference trajectory. The majority of swingby opportunities occur in the 530 - 675 day regime. Although most swingby opportunities have a longer trip time than the fast trip non-swingby opportunities, they offer a reasonably low IMLEO for all mission opportunities. An important point to note is that some years contain an outbound swingby opportunity, while some years contain an inbound swingby. the situation imposes less that an 18 month departure time between consecutive opportunities. This restrictive time frame could interfere with time restraints on the launch of HHLVs and assembly for the next mission.

Propulsion Option Swingby Opportunities

Swingby Opportunities for Expendable High Thrust Vehicles. 2010 - 2025



/STCAEM/brc/13Nov90

Chem/AB & NTR Vehicle IMLEO Comparision Data

Opposition Missions *with Venus swingby*

Year	Mass (t) Chem/AB		NTR - NERVA		NTR - Advanced		NTR -Adv Reusable
	15 % AB	30 % AB	T/W=20	T/W=3.5	T/W=20	T/W=3.5	T/W=20
2010	581	704	409	433	360	381	597
2013	601	728	363	388	323	345	588
2015							
2016	669	811	448	481	387	415	720
2017	532	644	410	434	360	381	580
2020	510	619	339	361	305	324	518
2022	638	773	450	478	391	415	659
2023	558	676	468	497	405	429	669

High Thrust Trajectory Assumptions for Propulsion Option Comparison

1) Maximum Earth, Mars arrivial Vhp = 7 km/s. When Mars aerocapture Vhp exceeds 7 km/s, a cryogenic propulsive burn using propellant stored in the TEI stage (Isp=475 s) is done to slow down vehicle to Vhp=7 km/s before Mars aerocapture. For ECCV entries at Earth where Vhp exceeds 9.7 km/s, a separate cryogenic propulsive stage (on ECCV) slows capsule down to Vhp= 9.7 before entry.

2) Mars Parking Orbit - 250 km periapsis altitude by 1 sol period

3) Correction for DLA Loss at Earth is performed at maximum apoapsis altitude during a 3-burn departure maneuver.

4) g-loss not accounted for

5) Corrections for DLA and apsidal misalignment at Mars will be performed at the optimal true anomaly, inclination, and period; note that an inclination will be chosen that allows for daylight landing.

5) No analysis will be performed at this time to evaluate if deep space burn maneuvers are the optimal correction for Mars departure DLA losses and Mars departure apsidal misalignment.

6) For 2025, deep space burn will be analyzed as a mode of minimizing IMLEO

7) Non Venus-Swingby cases will be analyzed for 350, 400, 450, and 500 day round trip times for 2018 & 2025 missions.

8) Midcourse corrections for legs w/o swingby = 50 m/s; with swingby = 100 m/s.

9) Aerocapture, expendable - optimize TMI and TEI Delta V's.

10) All propulsive, expendable - optimize TMI, MOC, and TEI delta V's.

ADVANCED CIVIL SPACE SYSTEMS

BOEING

Mission Delta V's and Departure Dates

Venus Swingby Cases

Δ = diff between arr Vhp & required arr Vhp limit

departure & arr dates — *Vhp limit at Mars capture=7 km/s, ECCV Earth entry Vhp limit=9.7 km/s*

	Year	Total trip time	Earth dep	Mars arr	Mars dep	Earth arr	(1) TEI dV	Outb Deep Space burn	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV
Oppositions	2010	673	*5529	5859	5889	6192	4.426	0	4.927	0	2.318	1.310	7.550	0	2.831
	2013	632	6618	6899	6929	7240	3.692	0	3.374	0	1.280	3.235	4.406	0	1.134
	2016	434	7463	7621	7651	6897	3.805	0	5.308	0	2.562	3.979	5.562	0	1.799
	2017	540	7850	8201	8231	8390	4.249	0	5.480	0	2.703	1.115	3.834	0	1.110
	2020	540	9055	9216	9246	9595	3.867	0	3.761	0	1.523	1.826	4.235	0	1.251
	2021	606	9518	9820	9850	**0124	4.258	0	5.795	0	2.944	2.520	8.172	0	3.138
	2023	586	**0194	0494	0524	0780	4.264	0	6.391	0	3.466	1.464	2.820	0	0.811
Conjunctions	2022	986	9811	**0167	0507	0797	3.882	0	2.820	0	1.012	1.168	3.611	0	1.037
	2020	860	9086	9196	9821	9946	4.792	0	5.656	0	2.823	2.996	6.164	0	2.087

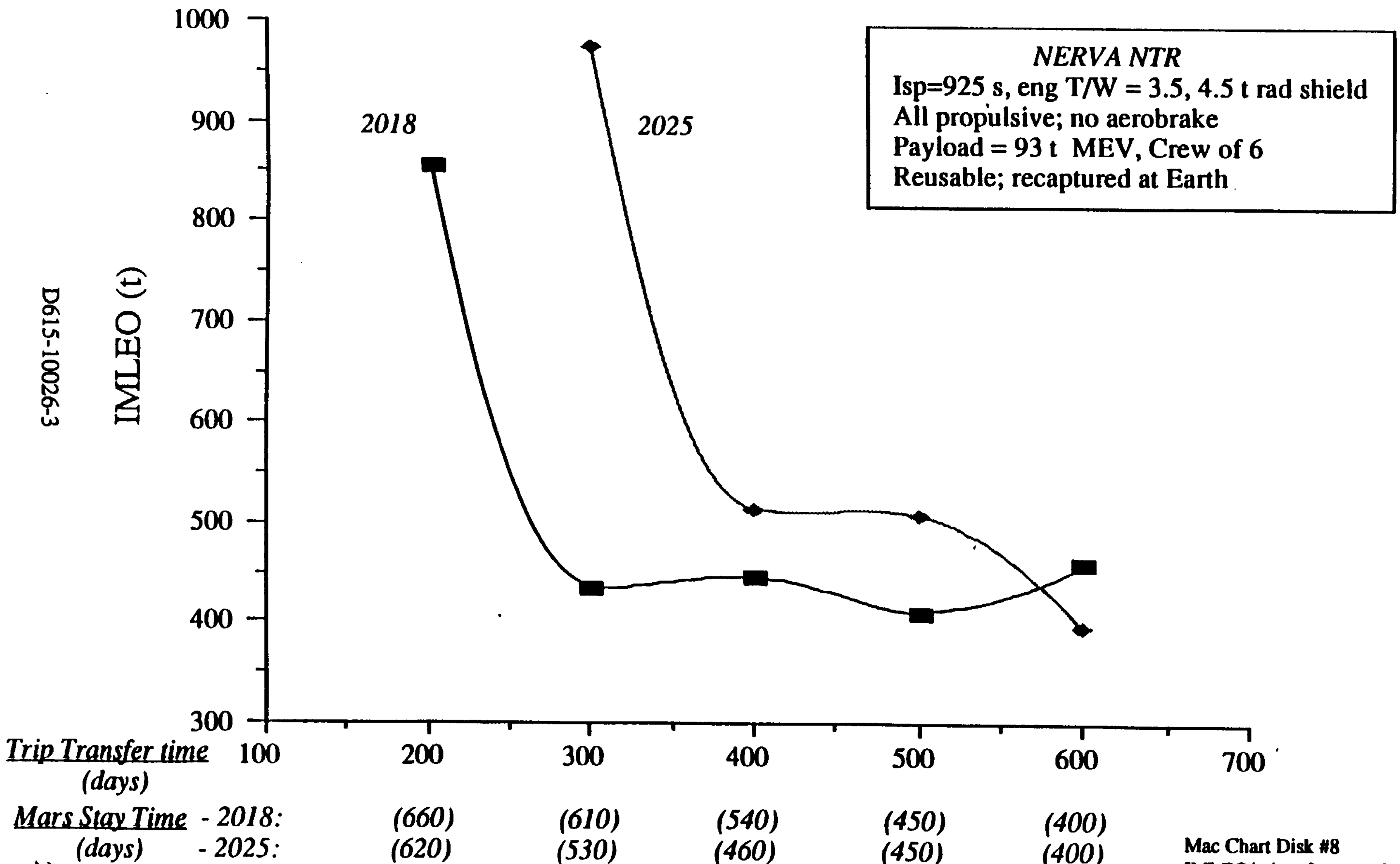
(1) *g-losses not accounted for* (2) all aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo chemical propulsion to slow veh down to Vhp=7 before aerocapture
(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo chemical propulsion to slow ECCV capsule down to Vhp=9.7 for entry

** 245xxxx -, ** 246xxxx Julian dates*

Stay time at Mars for all Oppositions missions = 30 days

disk #8/2010-2023 swby dVs

NERVA NTR Vehicle IMLEO vs Transfer time
Fast Transfer Conjunction Missions



Mac Chart Disk #8
IMLEO/t time fast t conj
Synthesis model run # ;377-386

ADVANCED CIVIL SPACE SYSTEMS

Mission Delta V's and Departure Dates-
Fast Transfer Conjunction Missions

For 2018 & 2025 Conj Missions

BOEING

dep & arr dates | Vhp limit at Mars capture=7 km/s, ECCV Earth entry Vhp limit=9.7 km/s; Δ = diff between arr Vhp & required arr Vhp limit

	Mars Stay time	Transfer trip time	Earth dep	Mars arr	Mars dep	Earth arr	(1) TEI dV	Outb Deep Space burn	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV
2018 'easy year'	660	200	*8300	8400	9060	9160	5.341	0	5.470	0	2.810	3.720	7.540	0	2.800
	610	300	8285	8435	9045	9195	4.132	0	3.480	0	1.390	1.960	3.590	0	1.030
	540	400	8270	8470	9010	9210	3.640	0	3.110	0	1.330	3.440	3.290	0	0.940
	450	500	8245	8465	8915	9195	3.908	0	3.020	0	1.170	2.240	3.140	0	0.900
	400	600	8270	8610	9010	9270	3.980	0	4.420	0	1.980	1.970	3.930	0	1.140
2025 'hard year'	620	200	**0645	0745	1405	1505	8.750	0	10.850	3.850	7.390	8.870	9.690	0.690	4.170
	530	300	0640	0796	1376	1521	5.730	0	5.410	0	2.630	5.300	6.370	0	2.190
	460	400	0621	0831	1331	1521	4.303	0	3.510	0	1.360	3.010	5.990	0	2.000
	450	500	0600	0860	1310	1550	4.783	0	2.900	0	1.110	3.250	3.960	0	1.150
	400	600	0575	0895	1295	1575	3.800	0	2.470	0	0.940	1.960	3.820	0	1.100

(1) *g-losses not accounted for* (2) all aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo chemical propulsion to slow veh down to Vhp=7 (km/s) for aerocapture
(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo chemical propulsion to slow ECCV capsule down to Vhp=9.7 for entry

*245xxxx -, ** 246xxxx Julian dates

disk #8/dVs 2018-2025 fast transf conj

Levied Requirements

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Nuclear Thermal Rocket (NTR) - System Requirements

During the coarse of the Space Transfer Concepts and Analysis for Exploration Missions contract (STCAEM), Boeing's Advanced Civil Space Systems group (ACSS) has conducted regular review meetings in order to define and derive requirements, conditions and assumptions for systems currently being developed.

As system definition and development progresses, technical experts provide documentation and rationale for requirements that have been derived. This real-time capturing prevents requirements and their associated rationale from slipping through the cracks. For example, a vehicle configurator may see the need for providing a minimum passage dimension for vehicle egress or ingress. This requirement would then be captured at an early development stage and would provide a history for the decision. This seemingly simple requirement may have large impacts on the design down the road and its traceability is important.

Derived requirements and rationale are later transfered to the Madison Research Corporation (MRC) where they are then entered into the system data base which has been developed for ACSS using ACIUS's 4th Dimension® software. The data base allows for easy access and traceability of requirements.

The charts that are contained within this document represent two collated copies of principal requirements and assumptions for February 2, and May 30, 1990. The systems defined include: (1) the Mars Transfer Vehicle (MTV), (2) Mars Excursion Vehicle (MEV), (3) Trans-Mars Injection Stage (TMIS), and the Earth Crew Capture Vehicle (ECCV). Each system is then broken down into subsystem headings of: (1) design integration, (2) guidance, navigation and control (GN&C), (3) electrical power, (4) man systems, (5) structure and mechanisms, (6) propulsion, (7) ECLSS, (8) and command and data handling (C&DH). The initials of each of the technical experts responsible for developing the supporting rationale for each of the requirements is indicated parenthetically next to each entry.

Although the majority of the derived requirements listed are directly applicable to all vehicles such as those powered by Nuclear Electric propulsion (NEP), Nuclear Thermal Rockets (NTR), Solar Electric propulsion (SEP) and reference Cryo, there are some that are not. Those requirements that are only directly applicable to a specific vehicle type are indicated within the entry. The italicized entries indicate a modification to an original requirement prior to the second revision of May 30, 1990.

Defining and re-examination of derived requirements will continue through the current contract.

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Derived Requirements

MTV Derived Requirements

- Design Integration
  - Two (2) communications satellites deployed in Mars orbit with total mass = 3000kg (GW)
  - Crew module must accommodate alternative advanced propulsion options (BD)
- GN&C
  - Capture trajectory entry interface for aerocapture not to exceed 6'g' limit and to preclude an uncontrolled skip-out (PB)
- Electrical Power
  - Solar power to be used for transfer phase, batteries to be utilized for sun occultation time while in Mars orbit (BC)
- Man Systems
  - Added protection to crew from Solar Proton Events (SPE) will incorporate use of a "storm shelter". (MA)
  - Consumables stored will suffice for crew residence time from 443-1018 days (includes abort), assumes 100% ECLSS closure of water and oxygen, 0% closure on food and .25 kg leakage per day (PB)
  - Two (2) astronauts able to pass through major circulation paths while wearing EVA suits. (SC)
  - Crew quarters shall provide sufficient volume for casual conversation between at least two (2) crew members (SC)

MTV Derived Requirements

(continued)

- **Man Systems** (continued)
  - Crew visibility during all maneuvers (docking/rendezvous) (SC)
  - There shall be 2 means of egress from each module for emergency escape (SC)
  - Crew module to accommodate 0'g' and induced 'g' environments (SC)

- **Structure and Mechanisms**
  - Airborne support equipment for aerobrake shall be 20% of aerobrake mass (PB)

MEV Derived Requirements

- Design Integration
  - Provide 15% of active weight for spares (JM)
  - MAV must be able to abort-to-orbit during descent phase (PB)
  - Twenty-five (25) ton down payload on manned vehicles (BS)
  - Protective covers provided for all mission critical systems (BS)

- GN&C
  - L/D range from 0.5 to 1.0 (GW)
  - Deorbit from 1 sol x 250 km periapsis orbit (GW)
  - Currently, cross range = ± 500km (GW)
  - Engine start before aerobrake drop (GW)
  - Approach path angle = 15° (GW)
  - Capture trajectory entry interface for MEV aerocapture at Mars not to exceed 6'g' limit on crew members and equipment and to preclude an uncontrolled skipout of the Mars atmosphere (PB)
  - Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1km cep and with beacon assuming 30m cep (PB)
  - Autonomous aerocapture capability at Mars, ~one (1) day before MTV (BS)
  - Aerobrake jettisoned in controlled manner during powered descent phase (BS)

MEV Derived Requirements

(Continued)

- Propulsion
  - Pre-descent checkout of engines to be provided (checkout extent TBD) (BD)
  - One (1) meter clearance established between engine bells and surface (SC)
- Electrical Power
  - Solar arrays to supply power to MEV following separation from MTV for fifty (50) day approach to Mars (BC)
  - Power for 50 day approach sequence to Mars shall be provided by solar arrays separate from the full MTV configuration. Arrays to be retracted 12 hours prior to Mars encounter, power shall be provided by batteries or other internal source (BC)
- ECLSS
  - Capability of two (2) crew cab represses (BD)
- Man Systems
  - Consumable s will suffice for a crew residence time of 30 - 600 days dependent on mission stay time and abort scenarios, assumes 100% ECLSS closure of water and oxygen, 0% closure of the food and .15 kg leakage per day (PB)
  - The maximum surface stay time is 600 days (PH)

MEV Derived Requirements
(Continued)

- **Structure and Mechanisms**
  - Shall be at least two (2) functionally independently pressurized areas for emergency conditions The shall be two (2) EVA suits stored in these areas (PB)
  - Establish 30cm clearance between all elements to allow for movement during high-stress maneuvers (SC)
  - Crew cab to have SSF diameter (4.4m), width (1.4m), and penetrations and attachments occur at rings. (SC)
  - Surface hab system to be removable later by surface construction transport vehicle and protected from damage by MAV blast during ascent start (BS)

MTV - TMIS Derived Requirements

- **Design Integration**
  - Flexible to support reference missions (interconnect design to support reference mission requirements (GW)
  - Fully modularized to utilize ETO capacity , the amount of modularization shall be a function of the ETO vehicle chosen (PB)
  - Assembly to be accomplished on-orbit, remotely and robotically (BS)

- **Propulsion**
  - Reference vehicle is launched "wet" with top-off (dry/wet issue to be traded) (JM)

- **Structure and Mechanisms**
  - Thrust structure - tanks - intertanks used as primary structure (GW)
  - The airborne support equipment mass for launch to Earth is assumed to be 7% for all hardware sets (PB)

Mars Transfer System Derived Requirements

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide (BS)
  - Equipment design life must account for mission duration plus one year (BS)
  - All components designed for 5 missions with refurbishment (except aerobrake) (BS)
  - Design for range of crew sizes, from 4 to 12 (BS)
  - L/D range from 0.5 to 1.0 for aerobrake vehicles at Mars (BS)

- **GN&C**
  - 8500 m/s maximum entry velocity at Mars (GW)
  - 100 m/s error-correction (post aerocapture) (GW)

- **Propulsion**
  - Engine out capabilities in all mission phases (BD)
  - Engine must continuously track C.G. of vehicle from beginning to end of all burns (BD)
  - Maximum gimbal angle of engines TBD (BD)

- **Man Systems**
  - Solar Proton Event (SPE) protection to be provided (MA)
  - Allow for direct viewing of all docking, berthing and landing procedures (SC)

Mars Transfer System Derived Requirements

(Continued)

- **Structure and Mechanisms**
  - All critical function lines and redundant systems shall run non-parallel (PB)
  - All systems shall function up to 2 years in a dormant state and having been subjected to the harsh space environment (PB)
  - The airborne support equipment mass for launch to Earth orbit shall be assumed to be 15% for all hardware except the aerobrake (PB)
  - Airborne support equipment mass assumption for the aerobrake shall be 20% of the aerobrake mass (PB)
  - Aerobrake will be launched to Earth orbit in sections for on-orbit assembly as the reference case (PB)
  - MTV and MEV aerobrakes have common layout of attach points (BS)
  - Vehicle elements will have removable debris shield panel cladding for protection during LEO operations. These panels will be removed and saved in LEO to be used for the next mission-opportunity. The panels will not add to the LEO debris environment (BS)
  - Mission vehicles will carry a robotic manipulation capability to inspect and maintain all exterior areas and systems (BS)
  - Structure optimized to minimize weight, operations, complexity and development effort (BS)
  - Greater than 30cm separation between all major vehicle exterior systems (i.e., tanks, modules) (BS)

- **C&DH**
  - Connectability between links maintained 90% of the time. Availability when scheduled - 98% connectability (PH)

MTV - ECCV Derived Requirements

- GN&C
  - Capture trajectory entry interface for ECCV aerocapture or aeroentry into Earth atmosphere not to exceed 6'g' limit on crew and personnel, and to preclude an uncontrolled skip out of Earth atmosphere (PB)
  - L/D = 0.25 (MF)
- Structure and Mechanisms
  - Interior materials must conform to NASA standards for outgassing, fire hazards,etc. (SC)

MTV - TMIS Derived Requirements

- **Design Integration**
  - Assembly to be minimized to extent practical. (KS)

- **Propulsion**
  - Passive thermal control system including zero-'g' thermodynamic vent system coupled to multiple vapor cooled shields. (JM)
  - TMIS insulating system is a continuously purged MLI over foam design optimized for minimum ground-hold, launch, and orbital boiloff. Includes vapor cooled shield (coupled to TVS) outside of foam. (JM)
  - TMIS tanks launched late in assembly sequence to minimize orbital stay time before TMI burn (, 6 months). (JM)
  - MTV tank insulation system is thick (2-4") MLI blankets. Multiple vapor cooled shields placed at optimum points in the MLI. (JM)

- **Structure and Mechanisms**
  - *Thrust structure - tanks - intertanks used as primary structure for cryo/aerobrake only* (GW)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

ADVANCED CIVIL SPACE SYSTEMS

Mars Transfer System Derived Requirements

BOEING

STCAEM/mha/30May9(

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide. The total wake closure angle is centered on the velocity vector. (BS)

- **GN&C**
  - *200 m/s error correction (post aerocapture)* (GW)

- **Propulsion**
  - *Engine out capabilities in all mission phases. NTR engine out capabilities TBD* (BD)
  - All passive cryogenic thermal control system.
  - No. MTV-TMIS fluid transfer before Earth departure. (MEV tanks refrigerated or filled after MOI)

- **Structure and Mechanisms**
  - Aerobrake externally mounted to vehicle for launch to Earth orbit ("Ninja Turtle" concept) (PB)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

MEV Derived Requirements

- Design Integration
  - Provide 15% of active weight for spares (JM)
  - MAV must be able to abort-to-orbit during descent phase (PB)
  - Twenty-five (25) ton down payload on manned vehicles (BS)
  - Protective covers provided for all mission critical systems (BS)

- GN&C
  - L/D range from 0.5 to 1.0 (GW)
  - Deorbit from 1 sol x 250 km periapsis orbit (nominal) (GW)
  - Currently, cross range = ± 500km (GW)
  - Engine start before aerobrake drop (GW)
  - Approach path angle = 15° (GW)
  - Capture trajectory entry interface for MEV aerocapture at Mars not to exceed 6'g' limit on crew members and equipment and to preclude an uncontrolled skipout of the Mars atmosphere (PB)
  - Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1km cep and with beacon assuming 30m cep (PB)
  - Aerobrake jettisoned in controlled manner during powered descent phase (BS)

MEV Derived Requirements

• Design Integration

- Down payload on manned vehicles
  - ~ 25 mt down payload for reference MEV (includes habitat module) (BD)
  - ~ 0.7 mt down payload for the 'Mini-MEV' (crew habitat is provided by the ascent/descent cab) (BD)

• GN&C

- *Currently, cross range = ± 1000 km for high L/D aerobrake* (GW)
- *Landing approach path angle = 15°* (GW)
- *Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1 km CEP and with beacon assuming 30 m CEP* (PB)

• Propulsion

- Engine out capabilities for ascent/descent stages (BD)
- Passive cryogenic storage system: MLI with vapor cooled shields (JM)
- Gravity field environment eliminates need for zero-'g' acquisition and venting. (JM)
- Vacuum jacketed ascent tanks for Mars boiloff reduction. (JM)
- MEV propellant transferred from MTV prior to descent. (JM)

• Electrical Power

- *Solar arrays to supply power following separation from MTV for ~ 50 day approach to Mars. Arrays to be retracted TBD hrs. prior to Mars descent (cryo/aerobrake).* (BC)
- Batteries or fuel cells to provide power for ascent and descent phases. (BC)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

MTV Derived Requirements

- **GN&C**
  - *Capture trajectory entry interface for aerocapture options not to exceed 6'g' limit and to preclude an uncontrolled skip-out* (MC)
  - Aerocapture exit errors not to exceed 0.25° inclination, RAAN, and ARCP, and a 0.1 hr period (MC)
  - GN&C requirement for advanced propulsion TBD: (MC)
    - ~ NTR - capture into planned orbit ± TBD
    - ~ EP (electric propulsion options) - TBD

- **Electrical Power**
  - *Solar power to be used for transfer phase, batteries or fuel cells to be utilized for sun occultation time while in Mars orbit except for NEP*. (BC)
  - NEP power derived from existing power system with a backup energy supply via fuel cells (BC)

- **Man Systems**
  - Volume per crew guidelines extrapolated from historical data (SC)
    - ~ Transfer hab = 112 m^3/crew
  - Two independant pressurized volumes for safety (SC)
  - Gravity condition emphasized to accommodate 0-'g' and 1-'g' and for surface commonality (SC)
  - 2.3 m standard ceiling height for psychological and locomotion (SC)

- **Structure and Mechanisms**
  - All penetrations occur in barrel section to minimize mass. (SC)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

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Guidelines and Assumptions

Mars Initiative - Assumptions

- Multi-impulse TMI and TEI is permitted, (engine restart)
  (e.g., three-burn departures acceptable for TMI to ease launch declination window problems [Level II])
- Cryogenic propulsion for Earth/Mars departures and Mars descent (cryogenic/aerobrake for Earth and Mars are selected as reference)
- Proven cryogenic storage technologies will be used
- Advanced propulsion technology options include NTR. SEP, NEP, and GCR
- MTV expendable on "difficult" opposition missions; return to Earth via ECCV
- TMIS expendable for reference system
- 100 ton cargo requirement (cargo mission) met by two (2) standard MEV's without ascent stages
- Maximum size surface payloads on piloted MEV: 6 m diameter and 13 m length

Contingency, Flexibility and Reserves

Flight Performance Requirements and Reserves

- 2% ΔV for Space Transfer Vehicles
- Add 2% for performance requirements uncertainties in selected instances
- Compute finite burn losses and add to impulsive requirements.
- Include delta V requirements for launch windows from LEO.

Dry Mass Contingency Allowances

- None for existing hardware
- None for consumables and impulse propellant
  - Consumables requirements shall include needed mission flexibility allowances.
  - Propellant reserves generated by flight performance reserves.
  - Use 2% of tank capacity for liquid and vapor unusable propellant; counts as inert mass.

- 5% on slightly modified hardware
- 15% on new design/known technology
- 15%-25% on new design/new technology, complex design, and poorly-understood requirements.

Payloads include flight support equipment

Manager's Reserve Policy, e.g. between launch vehicle capability and manifesting, TBD.

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III. Operating Modes and Options

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Reference

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NTR Operating Modes and Options

The following charts show the top-level operational sequence of events for the complete NTR mission profile. Options occur at the point of assembly (on or off Space Station for initial buildup, assembly and checkout operation from a Nuclear Safe Orbit (NSO)), outbound and inbound Venus swingbys, coast corrections and reconfiguring, and capturing the entire MTV on Earth return or an ECCV recovery only.

The NTR, in the baseline, will operate out of the LEO node co-orbiting with Space Station. It will not, generally, be necessary to operate from a NSO, as the NTR, even after the entire Mars flight ,will build up only about 250 grams of fission products. This will allow it to operate within 20 km of the Space Station from a radiological point of view. We are estimating the collision avoidance distance imposed by the structures to be 150 km, well beyond the NTR's radiation concerns. Having the NTR operate from the NSO will permit access to it from the Space Station at the rate of once a year when the orbits align. This will unduly restrict operations for no appreciable benefit .

The NTR will leave from orbit by doing one to three TMI burns to escape, whatever is dictated by the need to attain the Declination Launch Asymptote (DLA) required for Mars Transfer. The expendable drop tanks will be jettisoned as they become empty to reduce the mass, and therefore the inertia that must be changed with succeeding delta Vs that must be performed.

The on-line self-check capability of the systems and subsystems will be used throughout the mission to monitor the vehicle health and indicate preventative maintenance. Due to the length of the mission (1-3 years) the vehicle must be self sufficient and capable of maintenance and repair with a limited crew (4-7 people). The length of mission time and the distance will impose limits on the communications and control of the vehicle that can be done by ground operations; the crew are on their own resources.

The NTR is an all propulsive vehicle and requires no aerobraking to slow it for Mars Orbit Capture (MOC). It will use the Mars Excursion Vehicle (MEV) for aeroentry to the surface once the MTV has been parked in orbit. The lander will operate like the lander for the Cryo/Aerobrake mission; that is, after final site selection, it will aerobrake into the atmosphere until the brake is no longer useful , jettison the brake and land on the descent thrusters like Apollo. The MEV will have descent abort capability with the ascent section in the event of an emergency to obtain orbit and be picked up by the MTV. of the lander

Once on the surface, the MEV establishes contact with both the automated MTV and Earth., then proceeds to carry out the surface mission. When the surface mission is complete, the ascent section liftoff leaving the descent section of the lander and surface habitat behind. The ascent section attains orbit and docks with the MTV, the crew transfers with the return samples and all extraneous mass is jettisoned prior to the Trans-Earth -Injection Burn.

The inbound return transit proceeds like the outbound leg, with options in Venus swingby , coast maneuvers and transit flight configuration. On Earth return, the baseline option is to have the NTR capture into a 500 km by 20 km, 28.5 ° elliptical orbit . From there the crew and samples will be transfered to the Space Station by an LTV (Lunar Transfer Vehicle). After 30 to 60 days the NTR will return to a Space Station co-orbit for refurbishment . As an option the crew and samples may return by an ECCV direct to Earth with the MTV continuing on to a heliocentric orbit from which it is not recovered.

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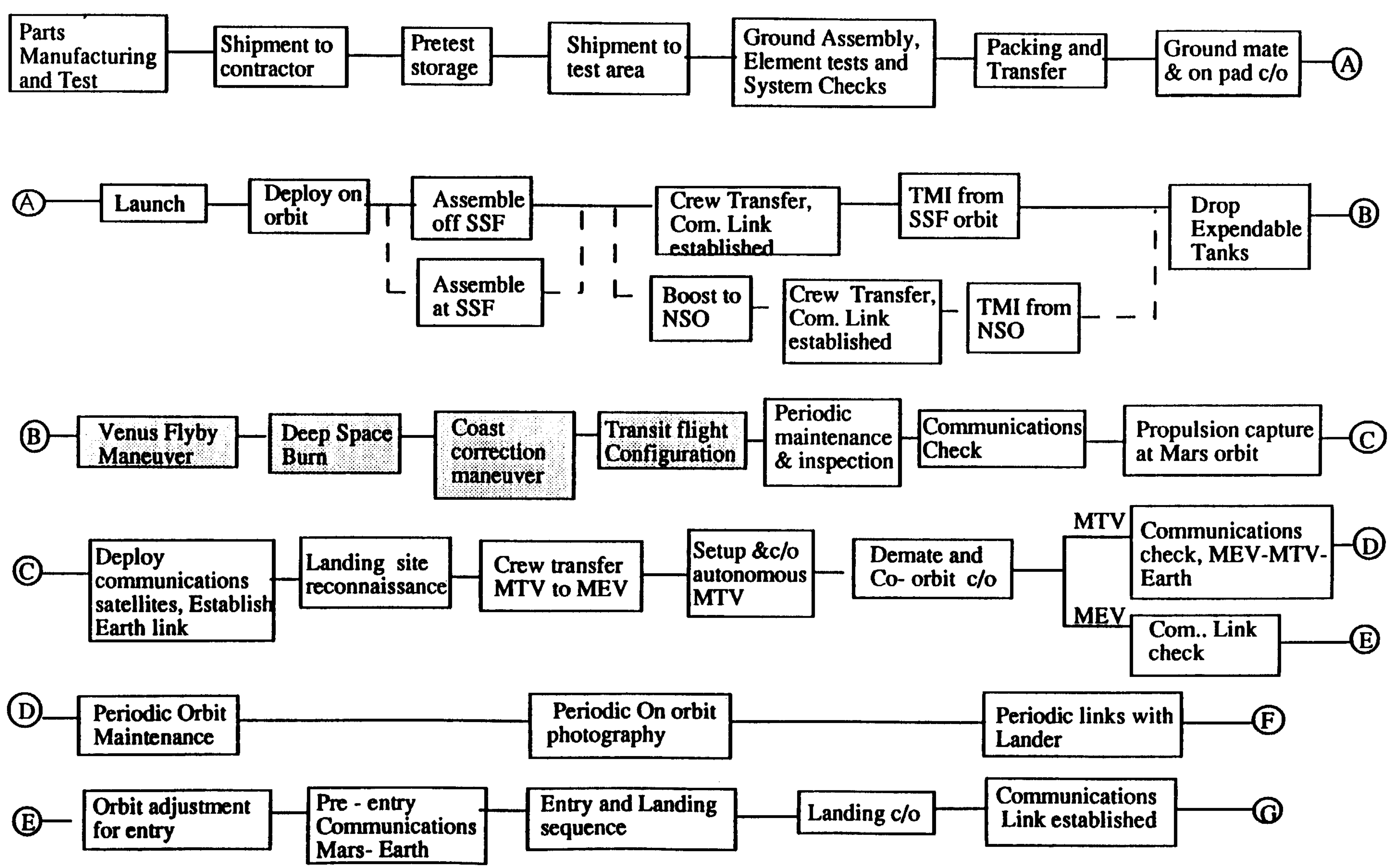
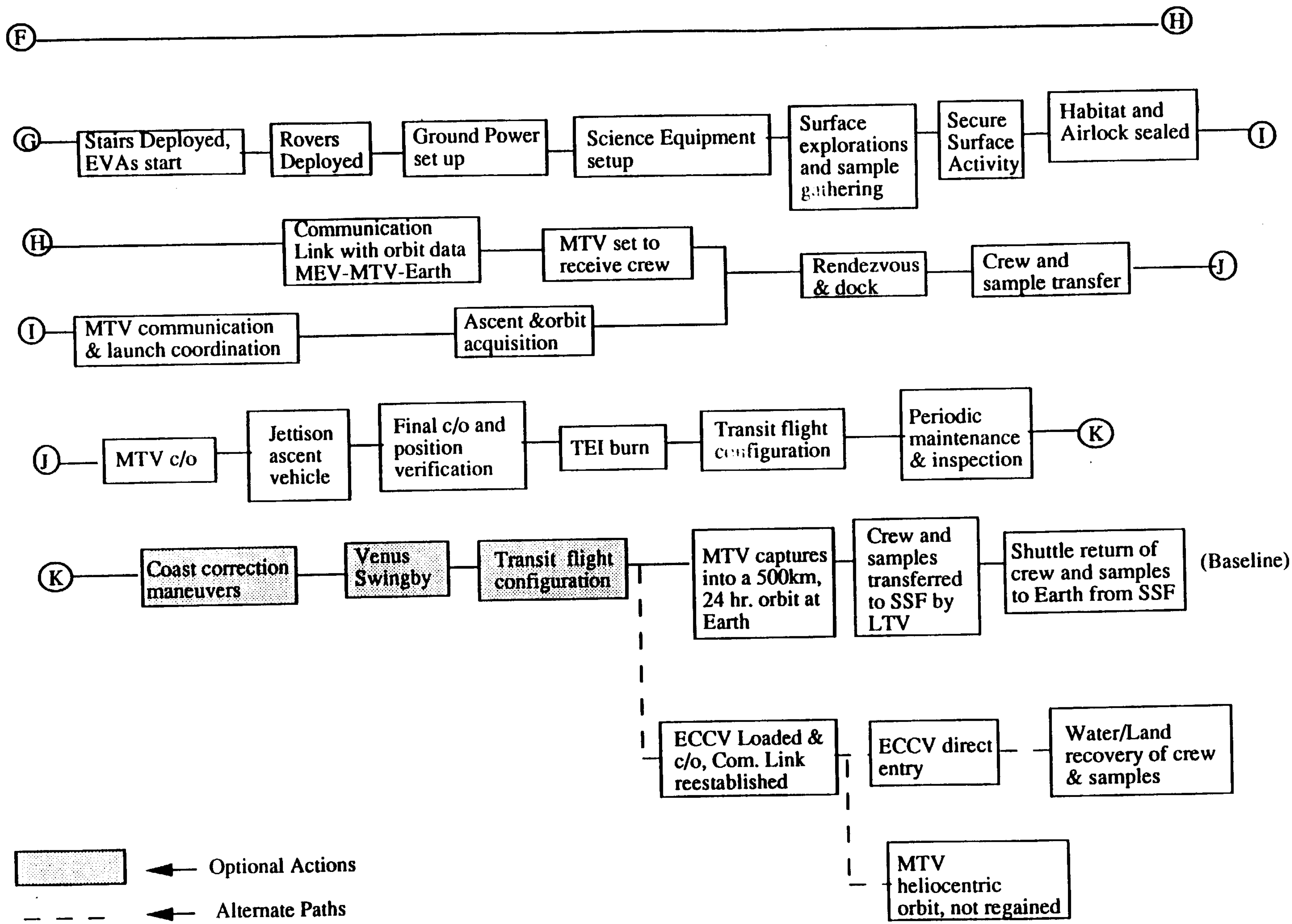
Mars Mission Operational Task Flow
Nuclear Thermal Rocket (NTR)





F
H
G
Stairs Deployed, EVAs start
Rovers Deployed
Ground Power set up
Science Equipment setup
Surface explorations and sample gathering
Secure Surface Activity
Habitat and Airlock sealed
I
H
Communication Link with orbit data MEV-MTV-Earth
MTV set to receive crew
Rendezvous & dock
Crew and sample transfer
J
I
MTV communication & launch coordination
Ascent &orbit acquisition
J
MTV c/o
Jettison ascent vehicle
Final c/o and position verification
TEI burn
Transit flight configuration
Periodic maintenance & inspection
K
K
Coast correction maneuvers
Venus Swingby
Transit flight configuration
MTV captures into a 500km, 24 hr. orbit at Earth
Crew and samples transferred to SSF by LTV
Shuttle return of crew and samples to Earth from SSF
(Baseline)
ECCV Loaded & c/o, Com. Link reestablished
ECCV direct entry
Water/Land recovery of crew & samples
MTV heliocentric orbit, not regained
Optional Actions
Alternate Paths

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IV. System Description of the Vehicle

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Parts Description

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System Description

Part Descriptions

Nuclear Thermal Rocket (NTR) Evolution. The nuclear thermal rocket underwent considerable development and testing from 1955 to 1973. The Nuclear Engine for Rocket Vehicle Applications (NERVA) was developed to the point of detail drawings, which can serve as a starting point for a new NTR program. The alternatives include building the NERVA as designed, incorporating new materials into the design to operate at higher temperatures, or designing a new NTR engine. The additional cost of the latter choices must be compared to the cost savings from increased performance.

In addition to raising temperature, lowering engine operating pressure is expected to raise specific impulse, although the degree of improvement is in question. The improvement comes in two ways: using higher expansion ratios for the nozzle, and recombination of dissociated hydrogen. The particle bed engine concept uses small encapsulated fuel particles in the core. The large surface area of the particles leads to a high heat transfer rate, and thus to a high thrust-to-weight ratio.

NTR Sensitivities. The NTR sensitivity of Earth departure mass to specific impulse is in the range of 1.5 to 2.0 to 1. Increasing thrust-to-weight from 5 to 10 yields a 4% reduction in Earth departure mass, while a further increase from 10 to 40 yields another 3% reduction in departure mass.

NTR Performance. The next chart shows expected dissociation of hydrogen as a function of temperature and pressure. Dissociated hydrogen is advantageous in an NTR for two reasons. First, the average molecular weight of the propellant is lowered from 2 towards 1; and, second, the energy of recombination may help maintain gas temperature in the nozzle during expansion. Both factors contribute to higher gas velocity, and thus higher specific impulse. The following chart shows specific impulse as a function of chamber temperature and chamber pressure from two references. This dissociated may occur in the low pressure NTR, how much is to be determined

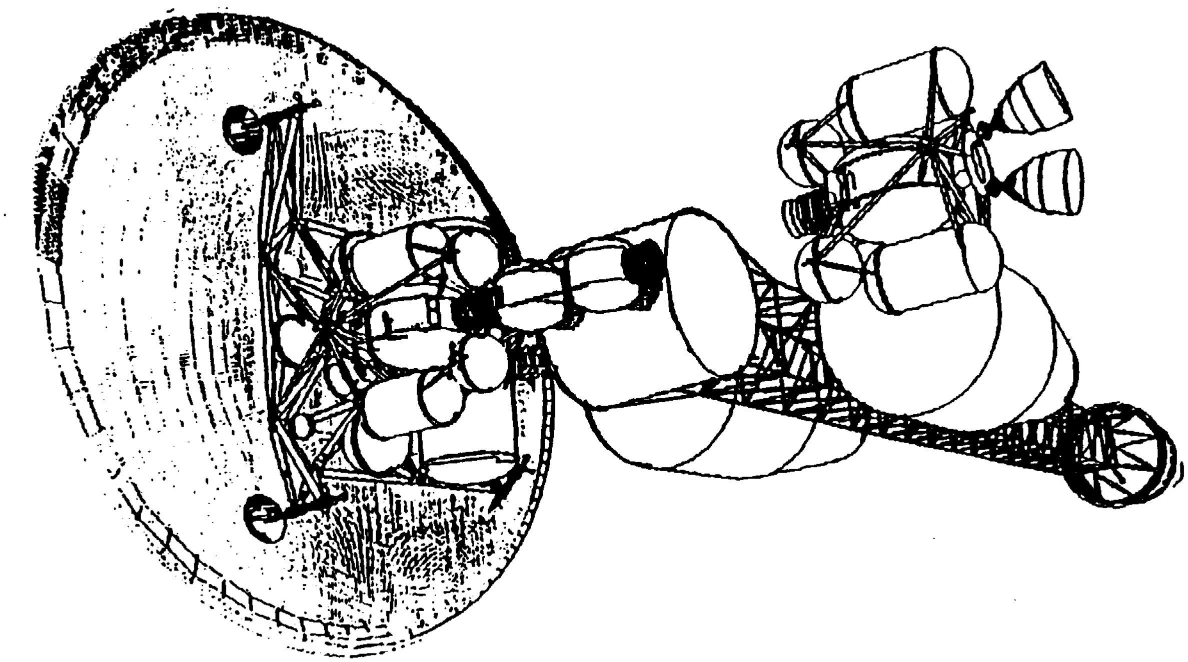
The following chart provides data on fuel temperature limits for different fuel compositions and endurance as a function of temperature for different types of fuel.

NTR Shielding. Shielding must be provided not only from direct radiation and particle emission from the reactor core, but also from secondary radiation. Secondary radiation is caused by the reactor bombarding parts of the NTR engine, such as the nozzle, which in turn become neutron activated or cascade generate additional particles. Thus the shield must be sized to cover all engine components on the reactor side from viewing by any of the spacecraft parts on the shielded side. Two charts show the configuration implications of this requirement.

Structural Trade. Consideration was given to combining the use of the propellant tanks as structural elements. It turns out that the tanks, if designed for one atmosphere internal pressure, are designed for a tensile load of 2 million pounds. This is far in excess of the NTR engine thrust, so keeping the tanks as structure imposes a mass penalty. The preferred alternative is to drop tanks and use a truss type structure.

Relative Development Effort Comparison. Estimates of the development effort for each propulsion element in a total Lunar/Mars program were made for various combinations of propulsion. The nuclear thermal rocket yielded the lowest effort estimate on a relative scale. This is only a gross comparison, not considering the differing cost of propulsion developments.

NTR Configuration



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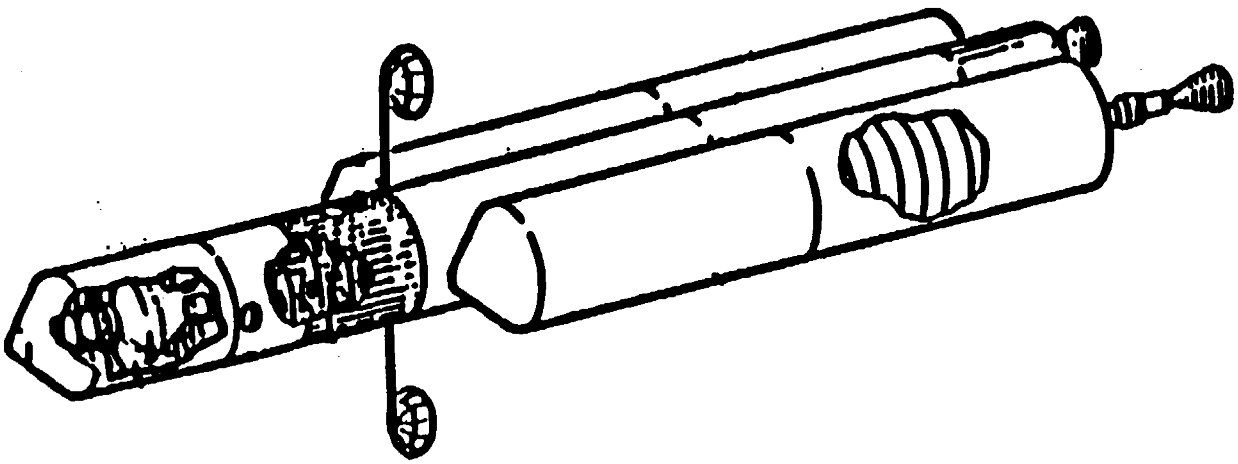
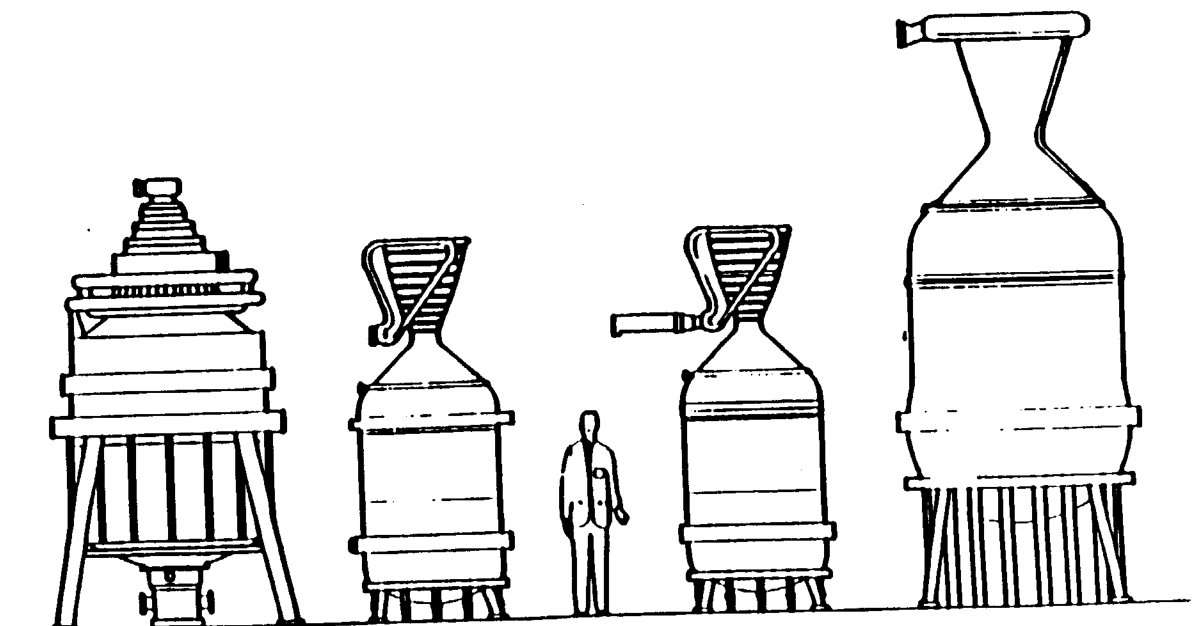
Early NTR Concepts

Initial nuclear rocket development was a joint Atomic Energy Commision-USAF project that started with exploratory research in 1953 bearing the name 'ROVER'. Soon to become a AEC-NASA program, early research was transformed into test hardware in July 1959 with the KIWI A test reactor (100 MWt, 0 lbf thrust) and peaking with the Phoebus 2A test reactor in 1967 (5000 MWt, 250k lbf thrust, 840 s Isp). In 1961, development began on the NERVA (Nuclear Engine for Rocket Vehicle Application) fight configuration series of full-up engines. Before termination of the NTR program in 1973, record performances of 62 minutes at continuous full power (NRX-A6), peak fuel temp 2750 K (PEWEE), peak fuel power density 5200 MW/m3 (PEWEE) and 28 single engine restarts (XE) capability, among other achievements of an extensive test program, were seen at the Jackass Flats, Nevada test range. A total of 20 reactors were designed, built and tested between 1955 and 1973 at a cost of approximately 1.4 billion before support for the program ended. Post-Apollo plans for manned expeditions to the planets were abandoned due both to major cuts in NASA's budget and its transition of focus to development of a space shuttle. In the manned Mars mission plans of the 1960's NTR propulsion was the system of choice. The vehicle sketch shown below is of a NERVA-powered Mars spacecraft presented to a US Senate committee by Dr Werner von Braun in August 1969. Much technical progress was realized in the areas of reactor reliability and safety, as well as in all phases of reactor/engine/component integration. The NERVA design specifications are retrievable, down to the actual subsystem component design drawings. Had the program continued past 1973, the next step would have been the development and testing of a flight qualified engine, with a most probable application as an upper stage propulsion system for the Saturn launch vehicle. Recent technology advances since the early 70's, especially in the areas of fuel element/coatings materials improvements and fabrication techniques would provide significant performance gains without the need for reactor redesign. Applications of the NERVA type engines for Mars missions vehicles is presented in the following charts. An important emphasis of this section of the study has been laid on validating the performance gain that can be expected for 1990-2010 technology NERVA derivatives.

Early NTR Concepts





70'S NERVA ENGINE

The sketches below are of the NRX class NERVA engines used as a departure point in the NTR propulsion analysis and trades

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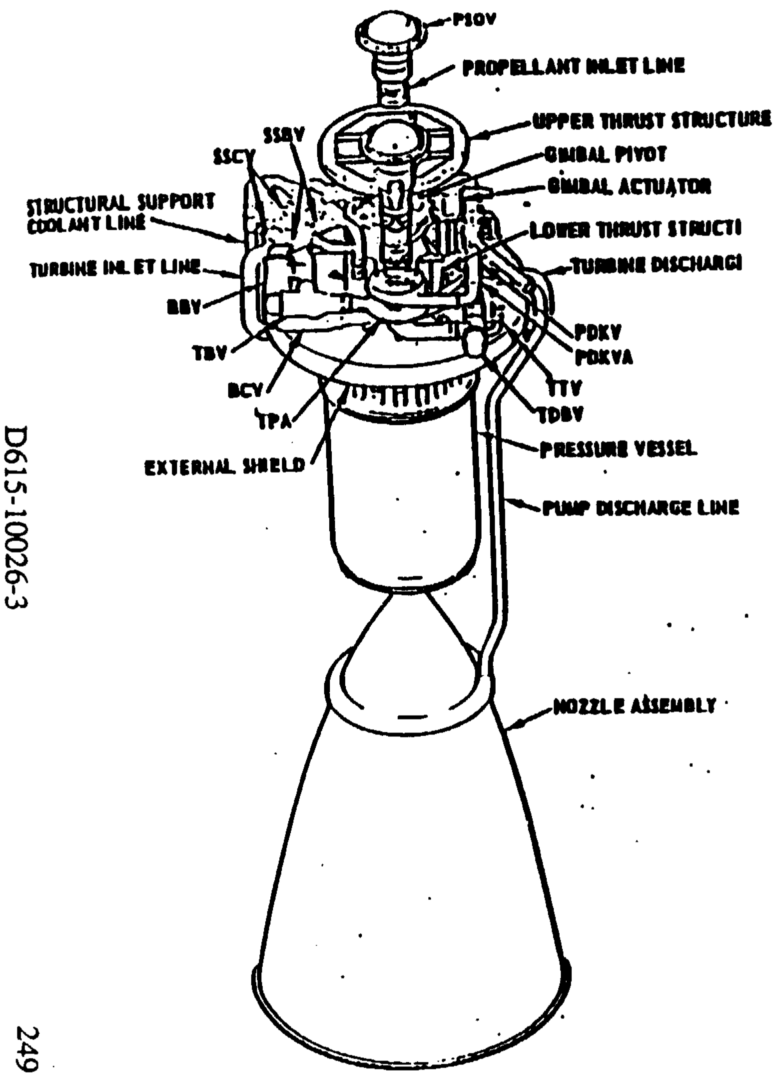
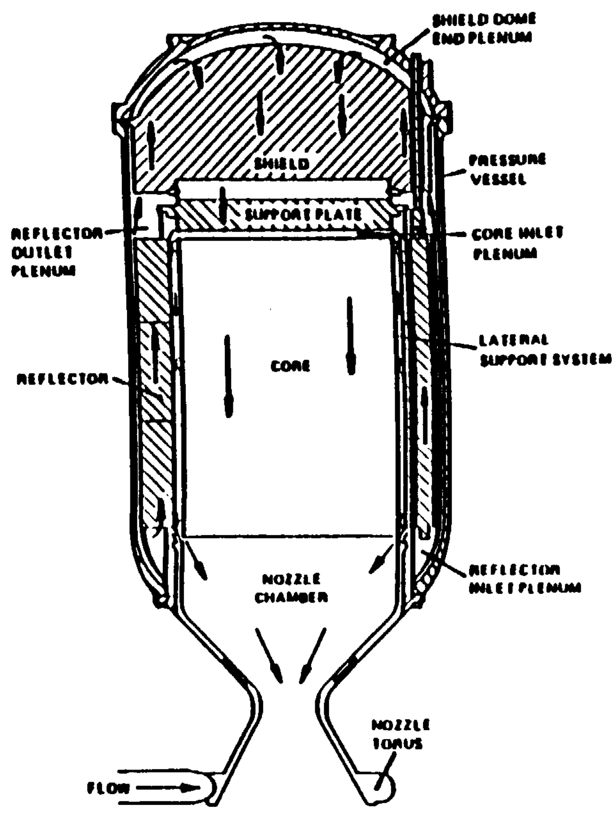
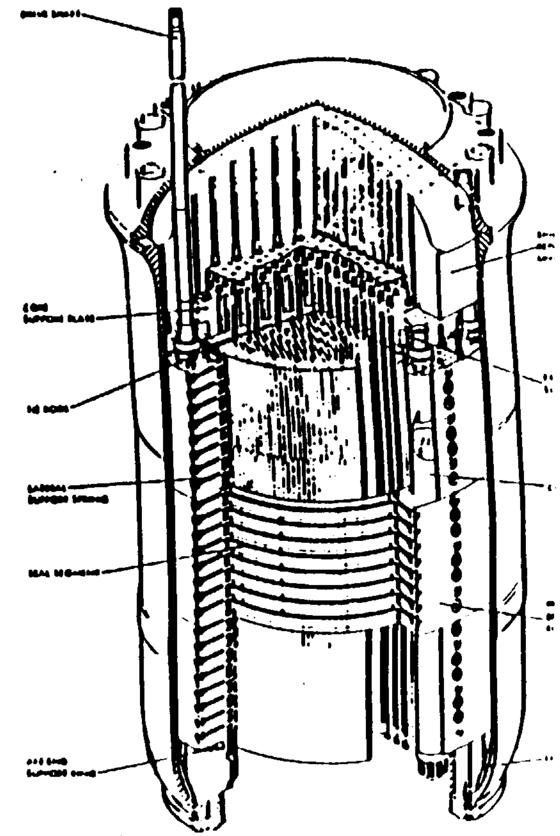
70's NERVA Engine

BOEING





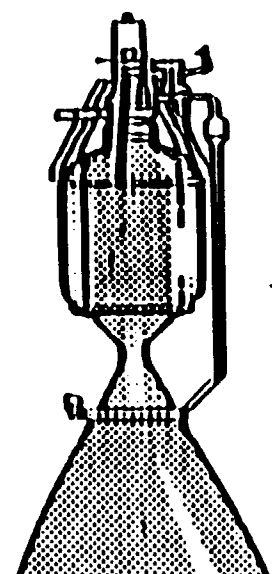
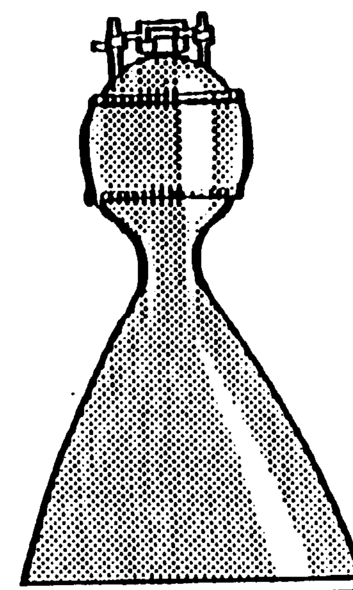


Advanced Propulsion System Characteristics

Two enhanced NERVA technology engine system characteristics are listed along with a radial flow low pressure reactor engine design for comparison with the demonstrated NERVA operating characteristics typical of the late 1960's.

Advanced Propulsion

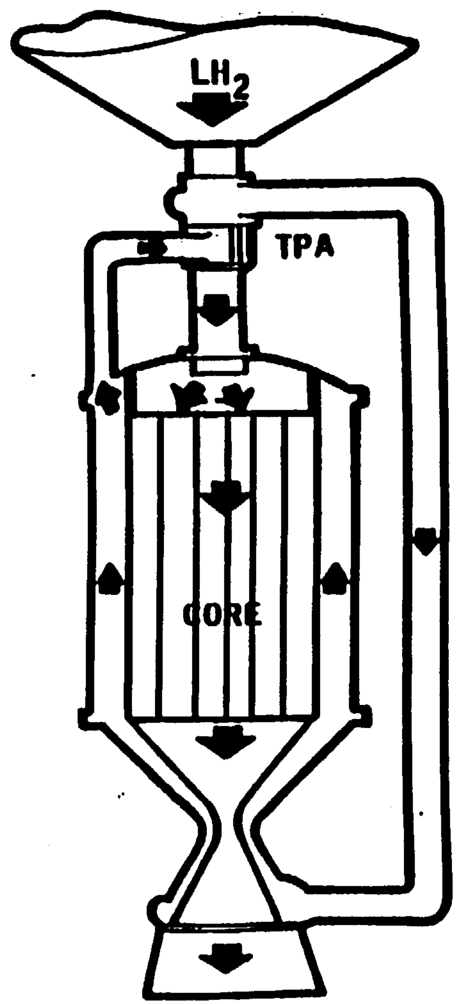
System Characteristics

System	Solid Core Nuclear Thermal			
	Axial Flow (NERVA)			Radial Flow
Engine	Demonstrated	Advanced	Low Pressure	Low Pressure
Thrust	75000 lbf	75000 lbf	1500 lbf	5000 lbf
Isp	850 sec	925 sec	1100 sec	1250 sec
Pressure	450 psi	450 psi	15 psi	7 psi
Temperature	2250 °K	3000 °K	3200 °K	3500 °K
T/W	3	6	0.5	2
Description				

Distinctions between Low Pressure NTR Reactor Concept and demonstrated NERVA Reactor

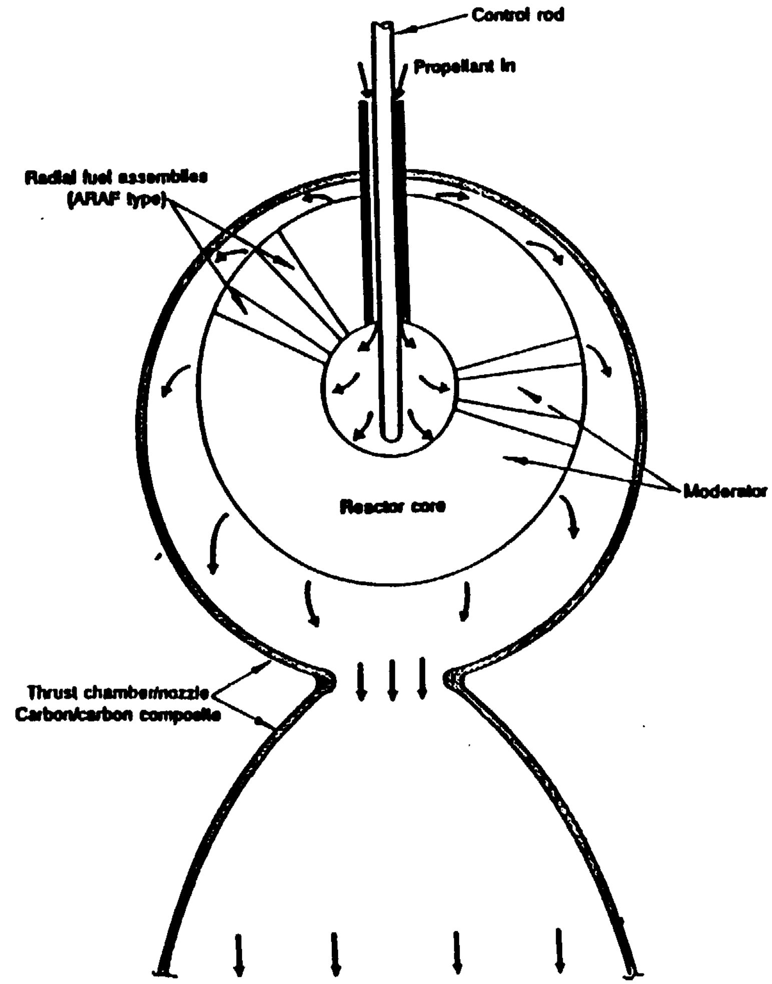
70's Axial Flow NERVA configuration

- High chamber pressures: 450 psia
- Pump feed engine; topping cycle shown in diagram



Radial outward flow Low Press reactor sys

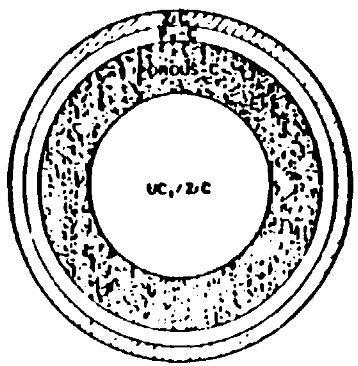
- Low chamber pressures: 10 psia and lower
- Pressure feed system



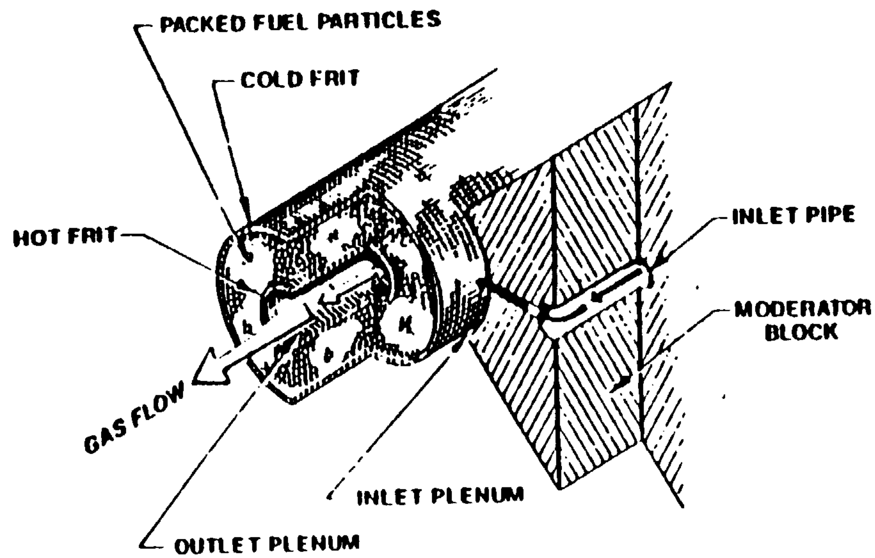
Particle Bed Reactor Concept

Characterized by High Engine T/W

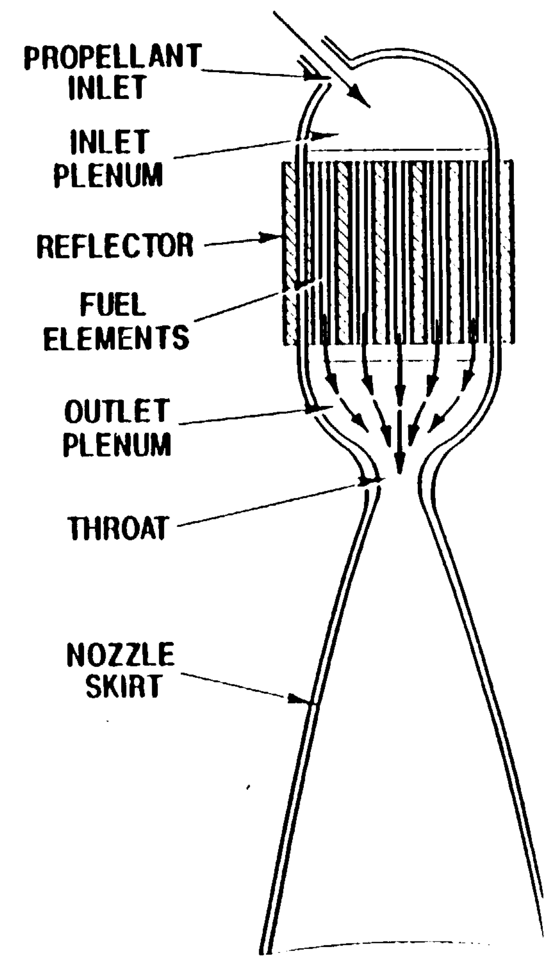
FUEL PARTICLE



BASELINE FUEL ELEMENT & MODERATOR BLOCK



ROCKET



2016 Advanced NERVA NTR Reference Vehicle Sensitivity to Isp and Engine T/W

The sensitivity of the reference NTR vehicle (IMLEO = 698 t) to changes to Isp is shown on the left with engine T/W held constant at 6 and 20.

The sensitivity of the reference NTR vehicle to changes in engine T/W is shown on the right with engine Isp held constant at 925 sec.

2016 Advanced NERVA NTR Reference Vehicle Sensitivity to Isp & Engine Trust to Weight Ratio

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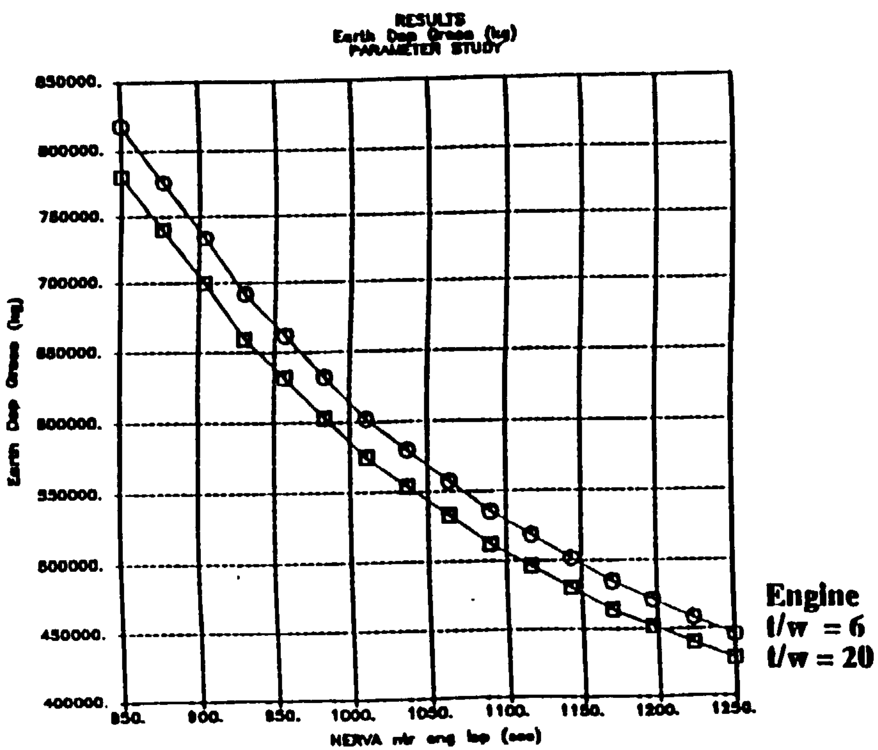
- 2016 Boeing #2 Modified Venus Swingby 464 day trajectory, propulsive capture at Earth, crew of 4
- Payload: MEV (77 t), MTV crew hab (32 t), 10.0 meter dia SiC/Al tanks - 14 % tank fraction
- One 75k lbf thrust eng, 4.5 t reactor shadow shield, eng wt & shield wt from NASA/LeRC propulsion task order

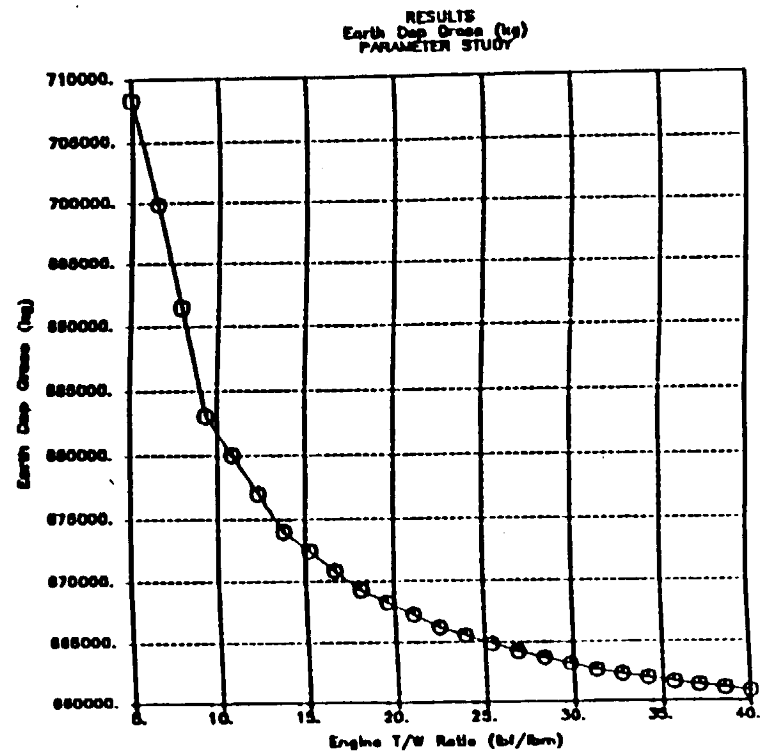
Vehicle IMLEO sensitivity to Isp

Top curve: Eng T/W held constant at 6:1, eng wt = 5669 (kg)
Bottom curve: Eng T/W held constant at 20:1, eng wt = 1700 (kg)



Vehicle IMLEO sensitivity to NTR Eng T/W

Engine Isp held constant at 925 sec



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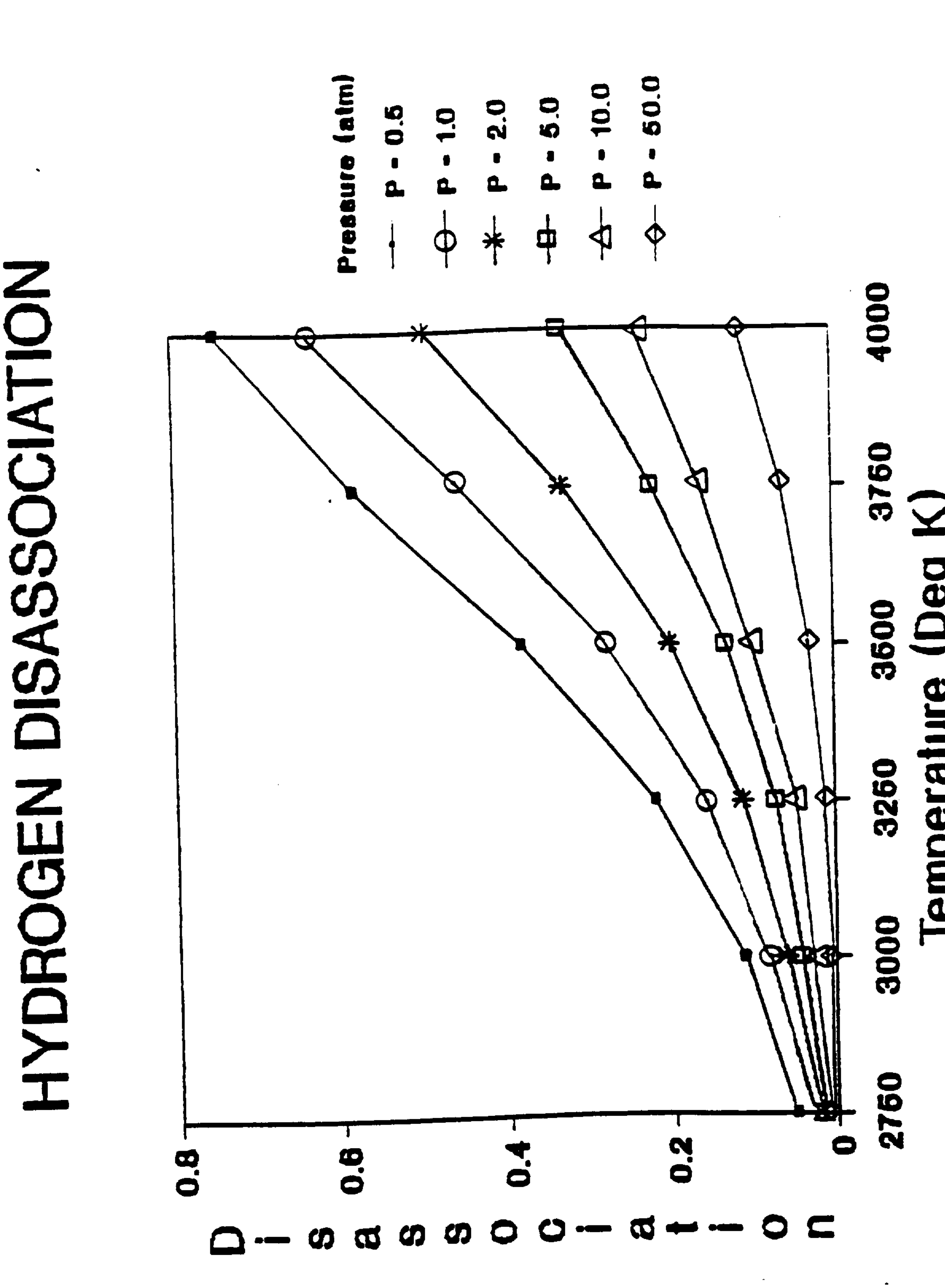


HYDROGEN DISASSOCIATION
Pressure (atm)
P - 0.5
P - 1.0
P - 2.0
P - 5.0
P - 10.0
P - 50.0
D i s a s s o c i a t i o n
0
0.2
0.4
0.6
0.8
2750
3000
3250
3500
3750
4000
Temperature (Deg K)

NTR Vehicle Isp as a Function of Hydrogen Temperature and Pressure

(1) Isp as a function of chamber temperature

Isp is proportional to the square root of chamber temperature. The figure on the left illustrates the rise in Isp with temperature for a range of 2600 to 5000 K. The UZrC ternary carbide fuel elements appear to have an upper operating temperature limit of around 3200-3300 K, given a choice of a operating temperature margin (typically about 278 K) from the melting point of 3590 K at a 40 % UC content. Once such a material limit has been reached, an additional gain in Isp can only be achieved by lowering the chamber pressure. The motivating force behind operating at lower pressures (and at these high temperatures) is the marked increase in the percentage of the H2 gas that disassociates into atomic hydrogen (see next chart entitled 'Hydrogen Disassociation'). Disassociation with accompanying recombination in the exhaust nozzle provides a significant increase in the Isp

A family of constant chamber pressure lines illustrate the theoretical gain in Isp that can be expected for lower pressure systems. These data is from a 1960 NASA report and are the basis for Idaho National Engineering Laboratories (INEL) analysis of conceptual low pressure reactor designs as a means for performance increases beyond NERVA derivatives. The theoretical Isp improvement, as indicated solely from this data, would be approximately 200 sec (1250 vs 1050 sec) for a 10 psia system vs a 450 psia system, if a 3200 K chamber temperature was maintained for both. The certainty of seeing this magnitude of improvement in actual practice is has yet unproven, and certainly has questions that might never be resolved until an actual reactor is tested - it would require a reactor design radically different from NERVA. Such a reactor concept specifically tailored to take advantage of disassociation at low pressures has been put forward by the INEL team; an illustration is given in the chart 'Distinctions Between Low Pressure NTR Reactor Concept and Demonstrated NERVA Reactor'.

(2) Isp as a function of chamber pressure

The same data as above, plotted vs chamber pressure on the x axis.

NTR Vehicle Isp as a fuction of Hydrogen Temperature and Chamber Pressure

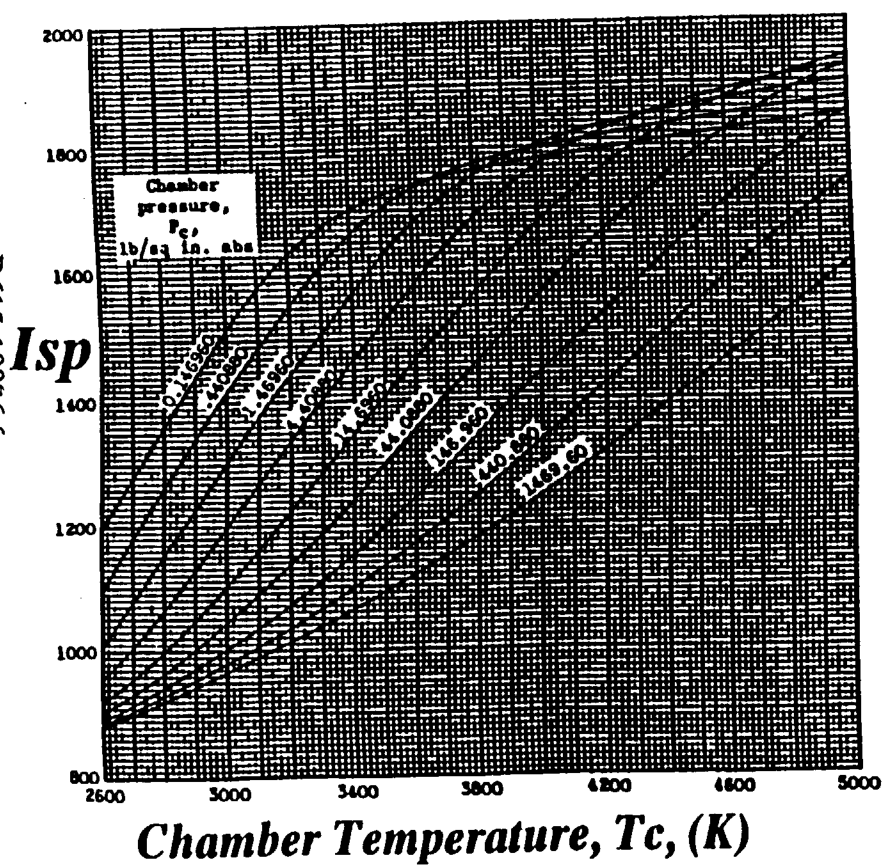
ADVANCED CIVIL SPACE SYSTEMS

BOEING

Isp as a fuction of chamber temperature

Source:

Compilation of Thermodynamic Properties, Transport Properties, and Theoretical Rocket Performance of Gaseous Hydrogen by Charles R. King, NASA/LeRC, NASA TN D-275 April 1960

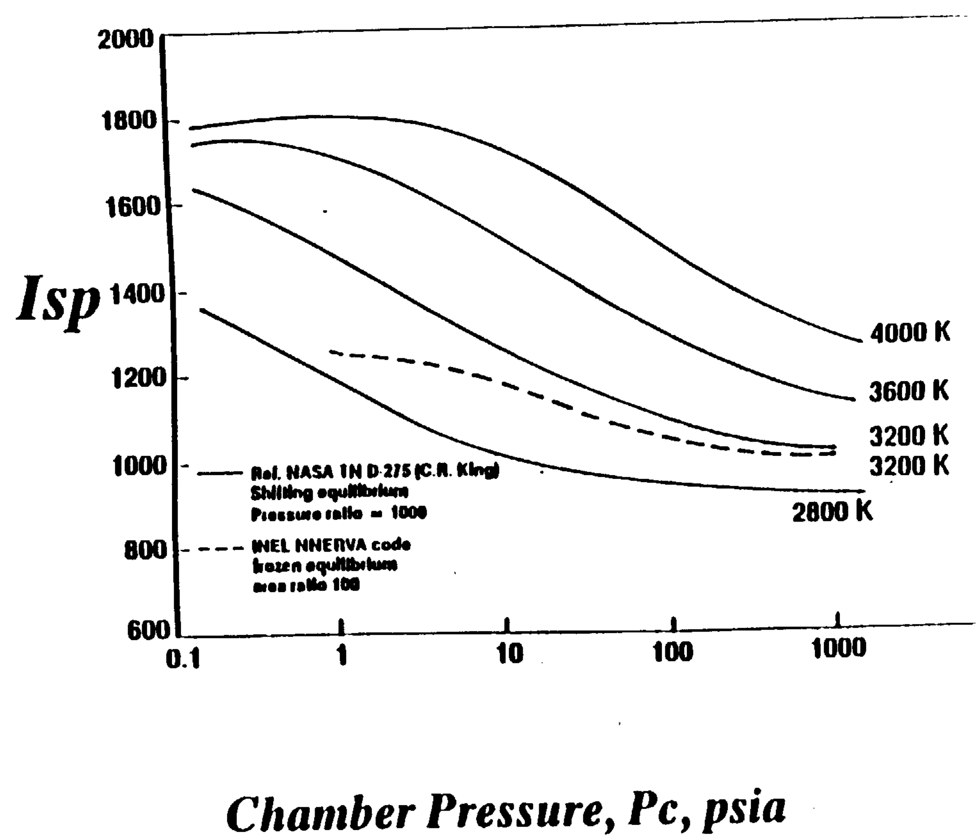


Isp in vacuum for gaseous normal hydrogen assuming equilibrium composition during an isentropic expansion to a pressure ratio of 1000

Isp as a fuction of chamber pressure

Source:

Pressure Fed Nuclear thermal Rockets for Space Missions briefing charts presented at NASA MSFC meeting 1989 J. Ramsthaler/C. Leyse, Idaho National Engineering Lab.



NTR Solid Core Fuel Element Temperature and Endurance Limits

(1) Fuel element temperature limits
ref: *Nuclear Space Propulsion*, H. F. Crouch, 1965

UZrC is the preferred ternary fuel for temperature and nuclear reasons. Its temperature advantage over that of UNbC (within the range of interest) is evident in the figure. The UTaC system is attractive at the less than 50 % UC content, but its neutron absorption cross section is disadvantageous. The melting point of UZrC ranges from a low of 5450 (F), to a high of 6100 (F)[3644 (K)]. For the selection of a 50% UC content [melting point~5850(F)/3505(K)], a typical reduction of 500(R)/278(K) (allowed to provide the required strength) would dictate an approximate maximum fuel element operation temperature of 5350(F)/3227(K)

(2) Fuel Element Endurance
ref: *Space Nuclear Power*, J.A.Angelo & D. Buden, 1985

The figure illustrates the anticipated lifetimes at various operating temperatures for graphite, composite and carbide fuel elements. If a ten hour life is desired, the reactor would have to operate around 2200-2300 K with a graphite matrix fuel, for composites approximately 2400 K, and for carbides possibly as high as 3000 K.

NTR Solid Core Fuel Element Temperature and Endurance Limits

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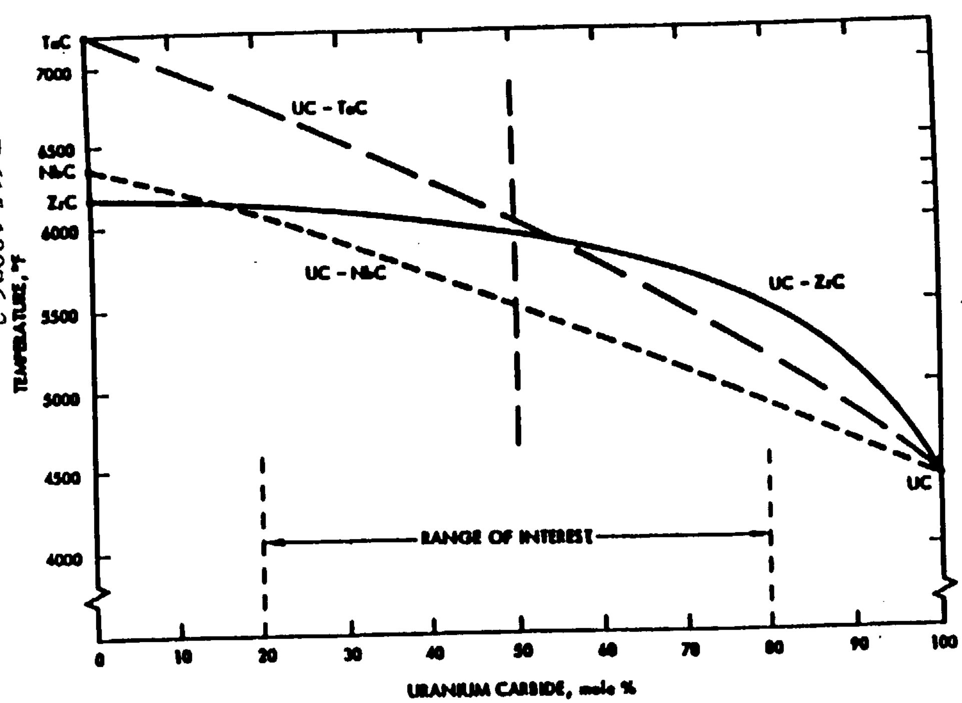
Fuel Element Temperature Limits for Carbides

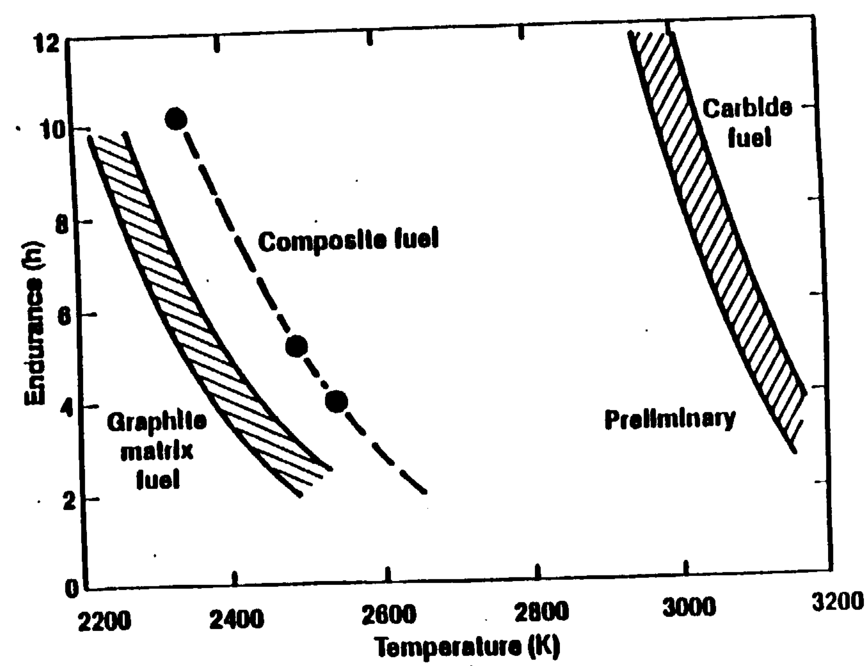
melting points of Ternary (=3 components in a solution) Carbide (=inorganic compound such as metal or ceramic with carbon) fuels vs UC mole % for 3 ternary systems which are satisfactorily stable with uranium:U-Ta-C,U-Nb-C, U-Zr-C;(Ta=tantalum,Nb=niobium,Zr=zirconium)



Source: *Nuclear Space Propulsion*, Holmes F. Crouch,1965

Fuel Element Endurance

Comparison of projected endurance of several fuels vs coolant exit temperature Graphite,Composite and Carbide fuels



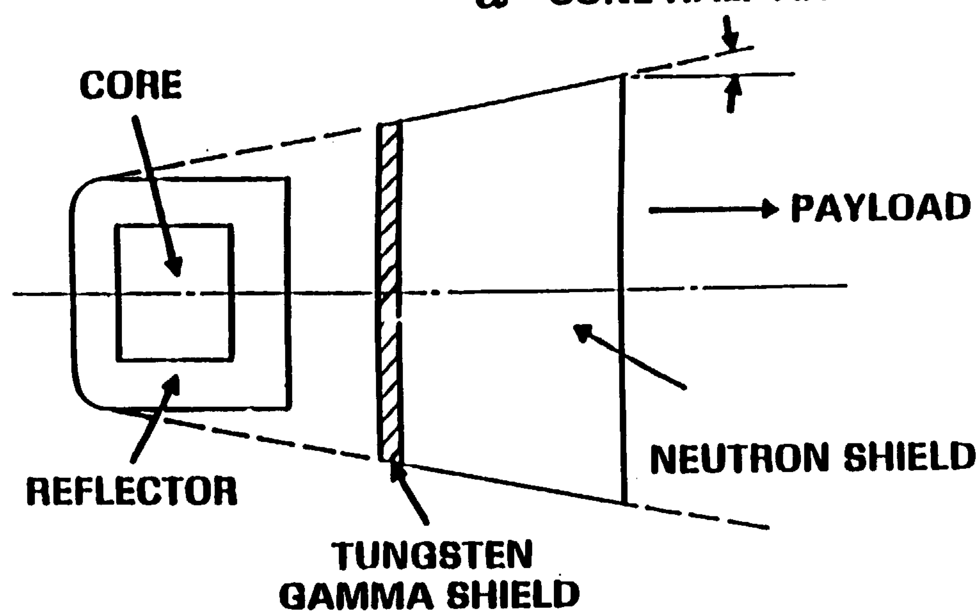
David S. Gabriel, *Statement to Committee on Aeronautics and Space Sciences*, US Senate, 1973

NTR shadow shield configuration

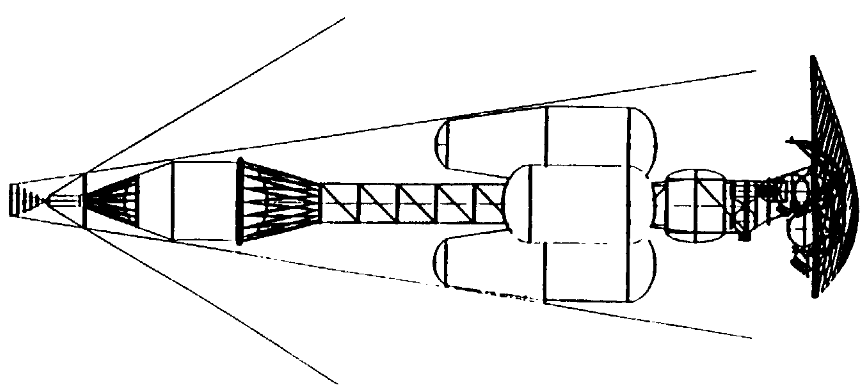
The NTR vehicle reactor shadow shield serves to shield the crew module and other structure (such as propellant tanks) from the high energy gamma radiation and the low energy thermal neutrons that are emitted from the reactor. A high density heavy metal material such as tungsten or lead serves to attenuate the gamma radiation while a material such as lithium hydride or water can be used to attenuate the thermal neutron flux. Minimizing the cone half-angle by configuration design is beneficial to minimizing the shadow shield size and weight.

NTR Shadow Shield Configuration

BOEING



Typical shadow shield configuration



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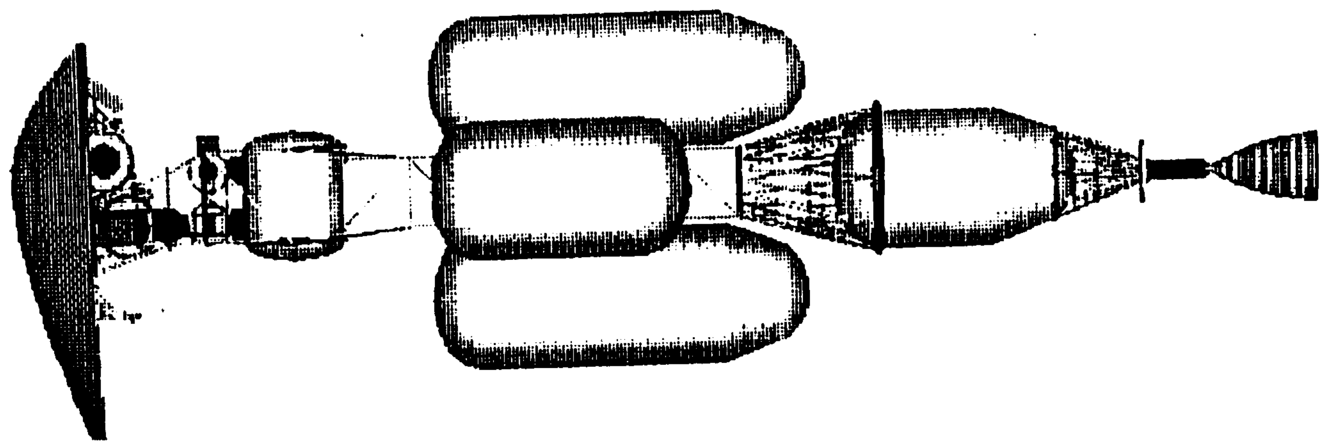
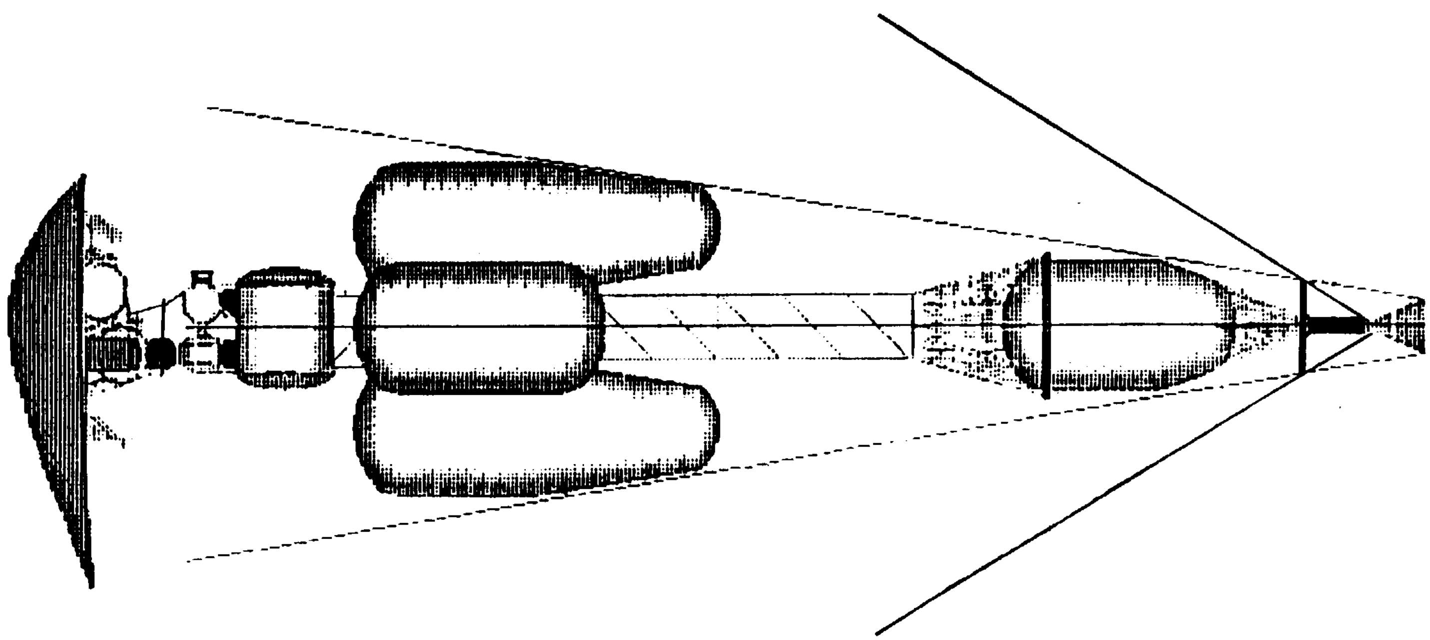
Radiation Assesment

Favors the Placement of all Tanks within Protected Cone Emanating from Nozzle Lip



Preliminary design



Updated design

Truss vs Tank for Structure Trade - 2016 Ref NTR Vehicle

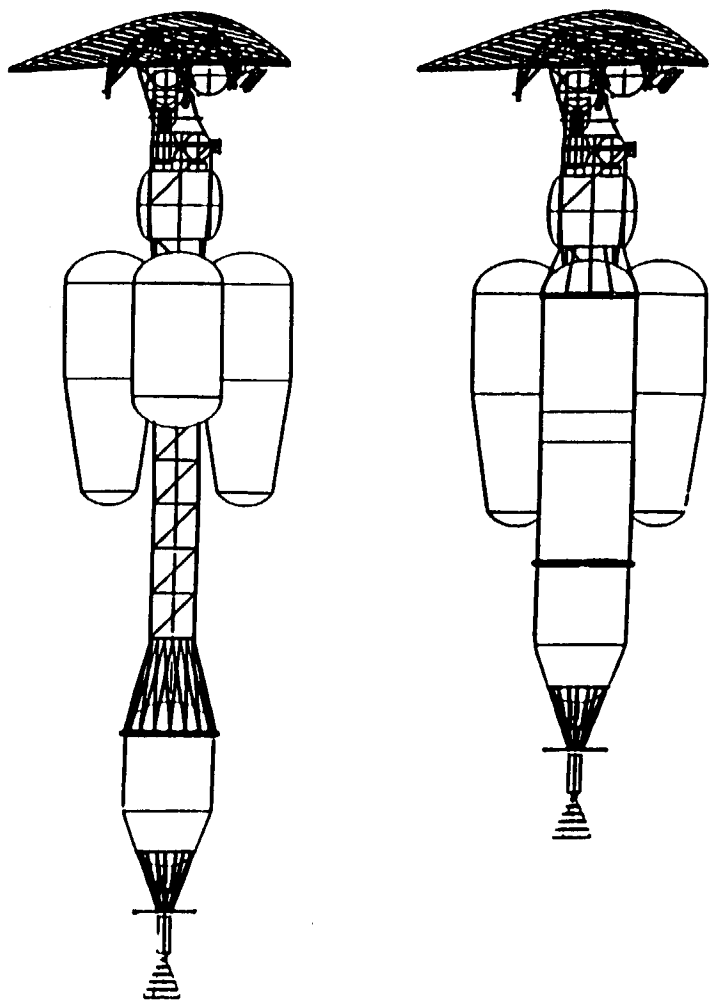
NTR vehicle configurations that utilize propellant tanks as structural members on the vehicle, instead of dropping them after they are emptied from a propulsive burn, entail on to the vehicle design a major weight penalty. The large hydrogen tanks typical for opposition class NTR Mars vehicles are covered with MLI, vapor cooled shields and meteor shields, and are relatively heavy. Tank fractions are 14% (tank fraction = tank wt/(total tank & propellant wt) or greater. It is a disadvantage for any NTR vehicle to have to carry a tank, emptied on the outbound leg, back to Earth, when it could have been dropped off earlier. A propellant tank serving also as a working member of the vehicle structure, could be replaced in that secondary capacity with a much lighter Space Station type truss. This would offer a significant IMLEO savings since the truss could be as much as an order of magnitude lighter than an empty MOC tank for one of these vehicles. A trade was done on the Boeing 2016 opposition mission reference vehicle design to determine the IMLEO penalty of keeping or dropping the large Mars orbit capture (MOC) tank.

(1) 2016 Reference Vehicle (truss system): The 2 MOC tanks were jettisoned immediately after the MOC burn. They did not serve as structural members - a SSF type truss served to connect the engine and aft tank to the crew habitat module. The large TMI and MOC tanks were attached to the truss. The weight estimate for the truss was 2400 kg using the standard SSF truss bays that weighted 160 kg per complete 5 m by 5 m bay. The two MOC tanks together weighted 25572 kgs empty, at a tank fraction of approximately 14%.

(2) 2016 Alternate Vehicle (no truss): The truss system structurally linking the engine and aft tank to the crew hab module has been replaced by using a single MOC tank as the connecting structure (see sketch). This tank can not be dropped after the MOC burn but must be carried back to Earth since its serves as a structural element. This single MOC tank weighs 25301 kgs, and must be kept pressurized inbound after it is emptied of its hydrogen at MOC.

Truss vs Tank for Structure Trade - 2016 Ref NTR Veh

keep or drop Mars Orbit Capture (MOC) tanks 6/8/90



Mac chart:truss vs tank structural trade

Vehicle synthesis model run #: marsntrmtv.dat;161,184

/STCAEM/bbd/8June90

Ref Vehicle:

- **Truss**: length 45 m (7 standard truss bays)
  width: 5 m by 5m SSF standard
  weight: 2400 kg

- **2 expendible MOC tanks**; empty wt: 25572 (kg)
  includes MLI,VCS, meteor shield, jetison hardware

- *Tot IMLEO of Reference Vehicle: 735,190 (kg)*

Operational concerns:
Provides radiation attenuating sep distance
Each of the 2 MOC tanks jetisonned after burn
2400 (kg) truss carried outbound & inbound

Alternate Vehicle:

- **Truss**: none; 0 (kg)

- **One 'In line' MOC tank** used as structural member
  empty wt: 25301 kg; includes MLI,VCS, meteor shield

- *Total IMLEO of Alternate Vehicle: 830,200 (kg)*

Operational concerns:
MOC 'inline' tank displaces truss & provides sep distance
25301 (kg) MOC tank carried outbound & inbound
95,010 (kg) IMLE0 wt penalty for carring empty tank
back to Earth

Truss vs Tank for Structure Trade - Results

The alternative vehicle design that utilized the MOC tank as a structural member was significantly heavier than the drop tank reference design. Having to carry the 25 ton of empty MOC tank on the inbound leg caused an increase in vehicle IMLEO of 95 tons over the IMLEO of the reference design with truss.

<u>Right hand plot</u>: indicates the increase in Trans Earth Injection (TEI) burn propellant necessary to inject the added inbound inert weight of this vehicle into the necessary Earth return trajectory. (The truss system of the reference vehicle at 2400 kg is approximately one tenth the weight of the alternate vehicles empty MOC tank used as structure). For a 25 ton MOC tank, approximately 22 tons of additional TEI propellant is required.

<u>Left hand plot</u>: This added propellant in turn increases the required outbound MOC and TMI propellant such that the overall penalty to vehicle IMLEO is about 95 tons.

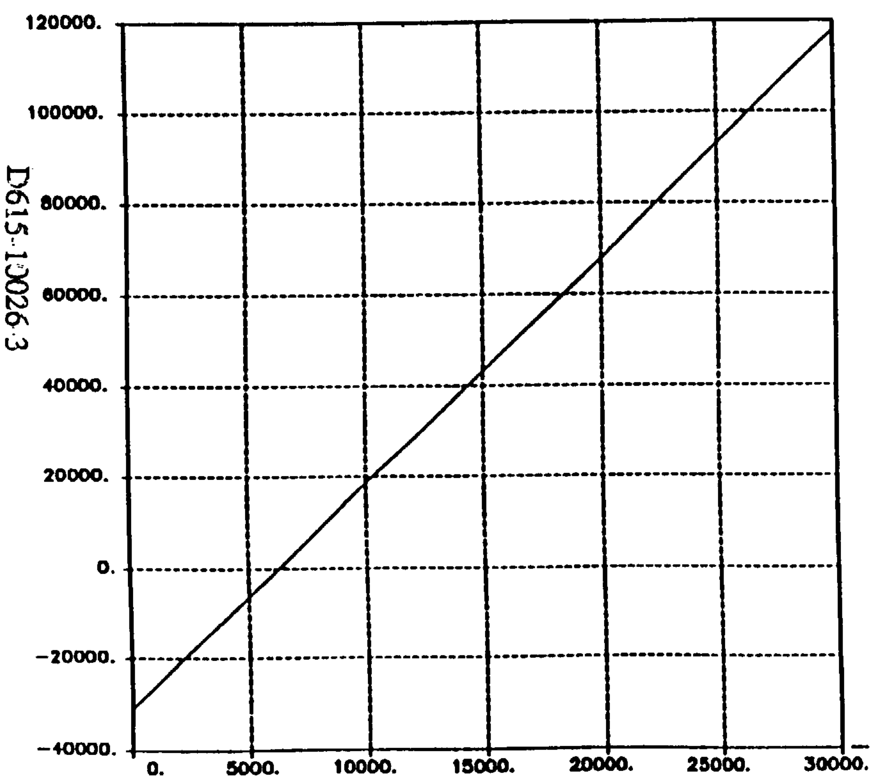
2016 Mission NTR Vehicle Truss vs Tank Trade Parametric Data

Veh mass delta's (between ref veh & 'tank return' alternate veh) vs MOC tank wt returned

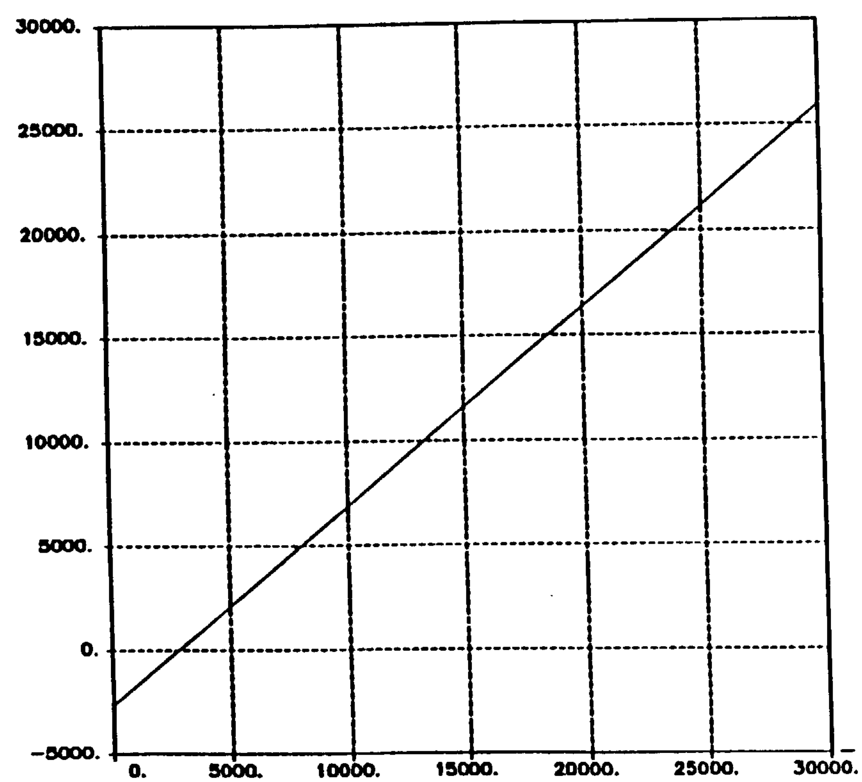
Vehicle Characteristics:

• MEV = 73118 kg •MTV mod= 34939 kg (crew of 4, 434 day trip) •NERVA eng wt = 9684 kg(Isp=925, eng t/w=3.5)
• Shadow shield wt = 4.5 t •Truss = 0 kg (tank displaces truss) • Mars orbit: 250 km by 1 sol • Four 10 m dia tank (14% t.f.)
•ECCV = 7 t • No Art-g • Storable RCS prop • vehicle departs LEO

Vehicle IMLEO delta
from ref veh IMLEO, kg



TEI burn prop delta
from ref veh TEI prop, kg



Mars orbit capture (MOC) tank wt returned to Earth, kg

Major Propulsion Element List for 2000-2030 SEI program

Primary Objective: Furnish a top level list of all major propulsion elements necessary to a 3 decade SEI total program entailing Lunar, Mars opposition (short stay) and Mars conjunction (long stay) missions.

Secondary objective: Considering four candidate vehicle combinations (differentiated by propulsion system choice, each of which might satisfy all the space transfer objectives of a comprehensive SEI program) roughly evaluate or 'score' the *total development effort required to bring each propulsion system element/technology up to flight readiness*. Having done so, sum all the element scores for each of the candidate vehicle combinations in order to ascertain which combination meets SEI program objectives with least overall propulsion systems development effort. The 4 candidates are listed below:

(1) Chemical Lunar with chemical Mars opposition (zero-g) & conjunction (art-g,tether system)
(2) NTR Lunar, NTR Mars opposition (zero-g) & Mars conj (art-g, vehicle rotation about its Cg, no tether)
(3) Chemical Lunar, NEP Mars opposition (zero-g) & NEP conj (art-g, tether system)
(4) Chemical Lunar, SEP Mars opposition (zero-g) & SEP conj (art-g, tether system)

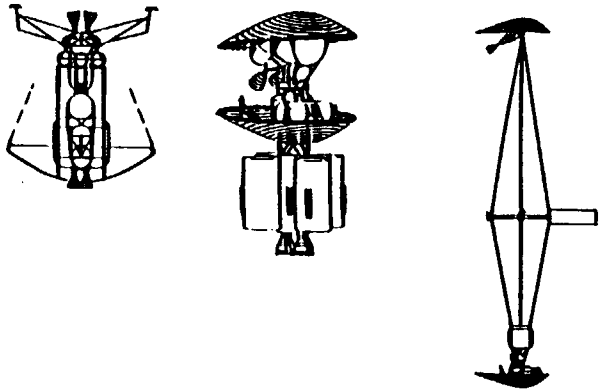
Scores: Primary list: the all NTR set had lowest total propulsion element count of 5, that is, 5 distinct elements were identified. The chemical/SEP combination followed with 8 elements, all chemical with 8, and chemical/NEP with 11. Differences in opinion as to what constitutes 'major' 'and 'distinct' propulsion elements might lead to slight variations in the totals, all depending on who does the counting.

Scores: Secondary list: The all NTR set scored the lowest in total propulsion elements development effort with a score of 13. the chemical/SEP combination and the all chemical set were about even with scores of 18 & 19 followed by chem/NEP at 27. These scores are relative, and only show how the 4 vehicle sets compare to one another; They are also subjective, and the differences in overall scores may be more pronounced, less pronounced or even change in rank depending on who is doing the evaluating. these rankings are not presented herein as the results of a precise technical trade study, but rather the results of a rough comparison 'methodology' with its major emphasis on a top down viewpoint in contrast to an analysis which as its emphasis on optimizing and/or selecting propulsion systems solely for individual missions.

Major Propulsion Element List for Specific Vehicle Sets
to Satisfy Lunar & Mars Objectives of 2000-2030 SEI Program

Moon	*zero-g* Mars *opposition*	*artificial-g* *conjunction*	*Propulsion Element*	*Development Effort Factor*

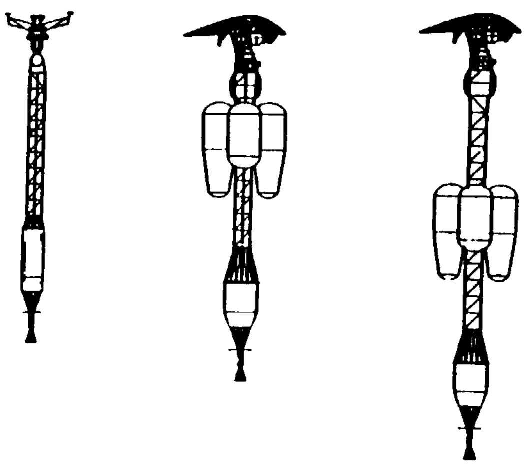
***Lunar*Chemical/*Mars*Chemical sys**



Lunar

1 LTV propul stg	2
2 LTV aerocapt brake	3
3 LEV propul stg	2
Mars zero-g vehicle	
4 MEV propul stg	2
5 MEV/MTV aerocapture brake	3
6 MTV propul stage	2
7 TMI propul stage	3
Mars artificial-g vehicle	
8 Art-g tether system	2
<u>*8 distinct propulsion elements*</u> *with development factor scores:*	**19**

***Lunar*NTR/*Mars*NTR sys**



Lunar

1 LEV propul stage	2
Common Lunar & Mars zero-g	
2 Common LTV/MTV NTR propul stage	6
3 Radiation handling/monitoring/shield	2
4 MEV propulsion stage	2
5 MEV descent heat shield	1
Mars artificial-g	
no necessary additions	
<u>*5 distinct propulsion elements*</u> *with development factors scores:*	**13**

Legend: (1) least development effort; (6) most development effort
Expected total resources that must be expended for such a propulsion element to acheive flight readiness

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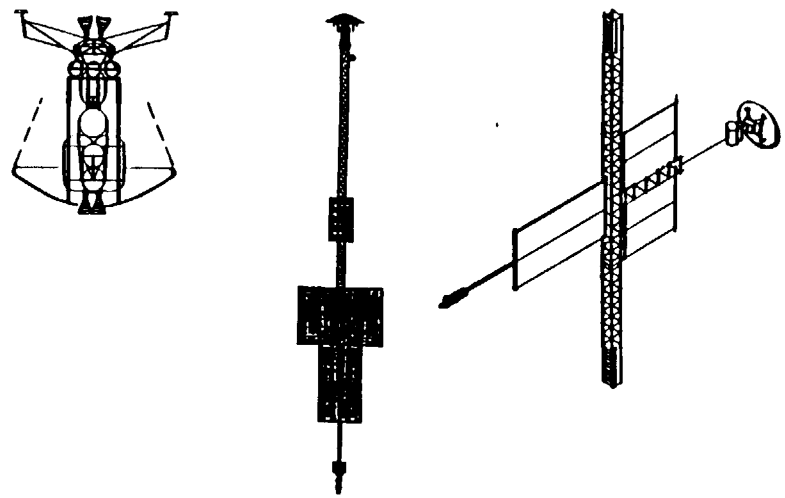
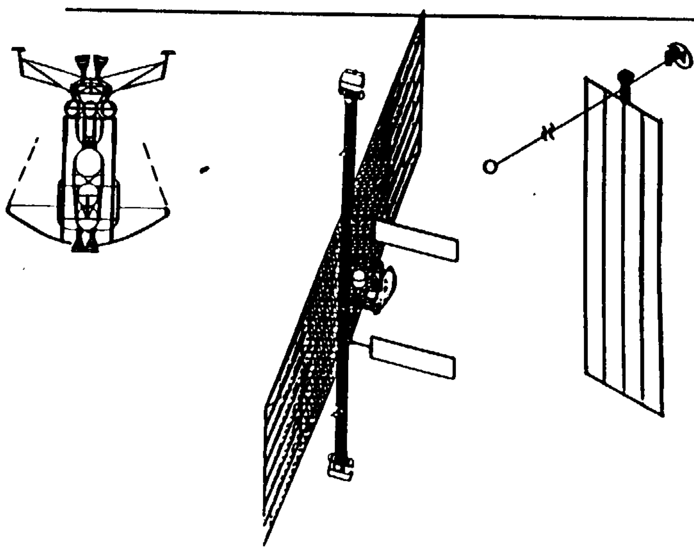
Major Propulsion Element List for Specific Vehicle Sets
to Satisfy Lunar & Mars Objectives of 2000-2030 SEI Program

ADVANCED CIVIL SPACE SYSTEMS — BOEING

Lunar	Mars zero-g opposition	Mars artificial-g conjunction



Propulsion Element	Development Effort Factor
Lunar	Lunar Chemical / Mars NEP
1 LTV propulsion stg	2
2 LTV aerocapture brake	3
3 LEV propulsion stg	2
Mars zero-g vehicle	
4 MEV propulsion stg	2
5 MEV descent heat shield	1
6 NEP reactor	6
7 Radiation handling/monitoring/shield	2
8 Dynamic conversion equip	2
9 Radiatiors	2
10 Electric thrustors	3
- Separate crew carrier to NEP 'spiral up' altitude	0 (use LTV)
Mars Artificial-g vehicle	
11 Artificial-g tether system	2
11 elements w sum of devel factors scoring:	**27**



Propulsion Element	Development Effort Factor
Lunar	Lunar Chemical / Mars SEP
1 LTV propulsion stg	2
2 LTV aerocapture brake	3
3 LEV propulsion stg	2
Mars zero-g vehicle	
4 MEV propulsion stg	2
5 MEV descent heat shield	1
6 SEP Solar array	3
7 Electric thrustors	3
- Separate crew carrier to SEP 'spiral up' altitude	0 (use LTV)
Mars Artifical-g vehicle	
8 Artificial-g tether system	2
8 elements w sum of devel factors scoring:	**18**

Legend:(1) least development effort; (6) most development effort
Expected total resources that must be expended for such a propulsion element to achieve flight readiness

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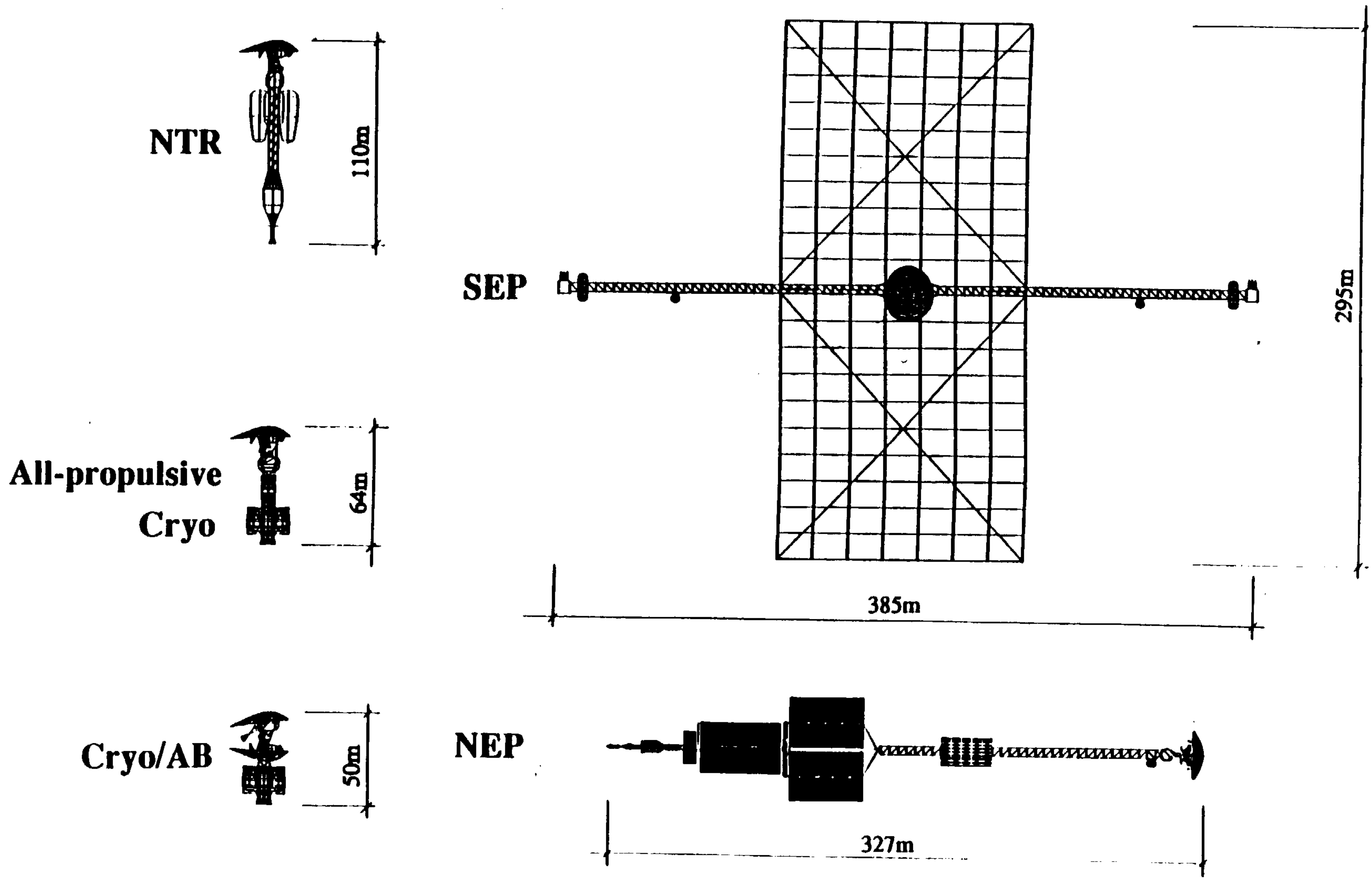
Reference Vehicles
Size Comparison



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Weights Statement

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Weight Statements

Summary and detailed weight estimates are provided for the Nuclear thermal rocket vehicle for the 2016 opposition mission opportunity. Assumptions made in the weight estimates include:

- Crew size of 4
- No Earth capture crew return vehicle
- Mission duration of 434 days.
- Improved technology (post-1990) for component weights (see technology

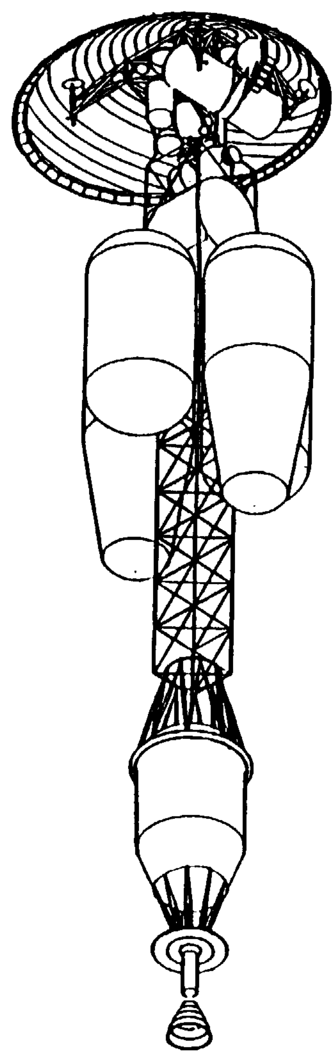
section). The reference mass for this mission case is 800 tons in low Earth orbit.

Mars NTR Reference Vehicle Weight Statement

The following six charts form a complete weight statement for the reference advanced NERVA NTR vehicle

Reference NTR Vehicle for 2016 Opposition Mission

Veh return to Earth for Reuse, no ECCV, Crew of 4, 434 day trip time Revision 5 5/22/90



Element	*NERVA T/W=3.5*	*PBR T/W=15*
MEV desc aerobrake	7000	7000
MEV ascent stage	22464	22464
MEV descent stage	18659	18659
MEV surface cargo	25000	25000
MEV total	**73118**	**73118**
MTV crew hab module 'dry'	28531	28531
MTV consummables & resupply	5408	5408
MTV science	1000	1000
MTV crew hab sys tot	**34939**	**34939**
MTV frame, propulsion, & shield wt	19777	12086
Earth Orbit Capture (EOC) prop	27756	24296
Trans Earth Inject (TEI) prop	59245	51727
EOC/TEI common tank wt	13845	12424
Mars Orbit Capture (MOC) prop	151680	138800
MOC tanks	25572	23962
Trans Mars Inject (TMI) prop	286146	262100
TMI tanks	43092	39973
ECCV	0	0
Cargo to Mars orbit only	0	0
IMLEO	**735190**	**673425**

all masses in kg's

Mac chart: M Ref NTR cover pg
synthesis model run# marsntrmtv.dat;161,183

Desc stage - MEV for 2016 Reference NTR Vehicle

Crew of 4, 4 adv eng's; Isp=475, 25 t surf cargo, descends from 250 km alt Rev 5 5/22/90

Group	Ref	Item	Fuel / Oxidizer	Notes
Desc stage inert	[98/99]	Single tank wt	225/117	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[124/125]	Meteoriod Shield	29/15	One 0.40 mm sheet of Al
	[122/123]	MLI	43/22	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	[126/127]	Vapor Cooled Shields	34/17	1 VCS at 2 x 0.13mm Al outer sheet w 0.57 kg/m2 honeycomb core
	[0/0]	Vacuum shell	0/0	not on desc tanks
	[2xi316]	Propel line wt	35/35	25 kg per tank + 10 kg for tank instrumentation
	[132/133]	Tank wt growth	41/23	15% wt growth
	[128/129]	Sum single tank inerts	407/229	Total single tank + tank inert wt
	[130/131]	***Tot: Fuel & Ox tanks:***	814/458	2 LH2 & 2 LO2 tanks
	[501]	Main propulsion	1127	4 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[102]	Asc frame & struc wt	562	4% of desc stage stg wt + 2% of surf crew mod mass
	[103]	Landing legs	1540	3% of total landed mass
	[i273,526]	RCS inert	428	Estimate from RCS prop load
	[135]	Propul, frame wt growth	493	15% of total inerts
	Sum	***Desc propul & frame inert***	4150	
Prop loads	[91+92]	Desc usable Prop	12061	*Desc propulsive veh dV= 931 (m/sec) from 250 km periapsis alt. by 1 sol orbit.*
	[0]	Desc boiloff	0	
	[101]	Desc RCS prop	1173	
	Sum	***Total Desc propellant load***	13234	N2O4/MMH prop, Isp=280 sec, *desc RCS dV=50 (m/sec)*
Aero brake wt		***MEV aerobrake:***		
		• Primary spar wt	1149	Structural design assumptions: 200ksi spar strength, 22.5 inch spar depth
		• Secondary spar wt	1200	NTR vehicle does propulsive braking of MTV & MEV into Mars orbit. This MEV
		• Honeycomb wt	3125	aerobrake is used only for descent to surface. It does not do aerocapture, which accounts
		• TPS wt	1526	for the wt difference between it and the MEV aerobrake wt (15138)for the Chem/AB veh
	[78]	Total:	7000	
	[77]	***Surface crew hab module***	25000	Level II Requirement: surf modulw, surf science & surf stay consumables
	[61]	***Asc veh total mass***	22462	from 'Asc stage' wt statement page
	[106]	***MEV mass***	73118	all masses in kg

synthesis model run#: marslander.dat;114

STCAEM/bbd/23May90 Mac chart: M Ref MEV decs veh wt-rationale

Asc stage - MEV for 2016 Reference NTR Vehicle

Crew of 4, 30 day stay, 2 adv eng's; Isp=475 , Ascends to 250 km alt *Revision 6 5/22/90*

Group	Ref	Item	Mass	Notes
Ascent Cab		Structure	998	SSF dia center cyl section w ellip ends. Stiffening rings added. See 'Structures pg'
		ECLSS	678	Open sys:CO2 adsorption unit, stored H2O,O2,N2, no airl., no hyg w. see 'ECLSS pg'
		Command/Control/Power	330	Power: fuel cells
		Man systems	82	Waste management sys/waste storage/medical equp.
		Spares & tools	192	Subsystem component level spares
		Wt growth	376	15% growth for dry mass
		Asc 'dry' mass	2656	Total cab dry mass
		Consumables (food & water)	62	Minimum; food and water only; 3 occupancy
		Crew/effects/EVA suits	760	Crew of 4, 100 kg EVA suit per crew member
	[22]	***Ascent cab gross mass***	3478	
			Fuel / Oxidizer	
Ascent stage inert	[45/46]	Single tank wt	301/138	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[68/69]	Meteoriod Shield	38/18	One 0.40 mm sheet of Al
	[50/51]	MLI	57/26	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	71/72/112/113	VCS & Vacuum shell	90/42	1 VCS and 1 Vac shell: both 2 x 0.13mm Al outer sheet w 0.57kg/m2 honeycomb core
	2x[i316+i318]	Propel line wt	35/35	25 kg per tank + 10 kg tank instrumentation
	[116/117]	Tank wt growth	57/28	15% wt growth
	[73/74]	Sum single tank inerts	578/287	Total tank & tank inert wt
	[114/115]	***Tot: H2 & O2 tanks:***	1156/574	2 LH2 & 2 LO2 tanks
	[500]	Main propulsion	564	3 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[118]	Asc frame & struc wt	469	3% of total asc stg propellant wt
	[i274,525]	RCS inert	222	Estimate from RCS prop load
	[54]	Propul, frame wt growth	188	15% of total inerts
	Sum	***Asc propul & frame inert***	1443	
Prop loads	[60]	Asc usable propellant	15482	*Asc veh dV= 5319 (m/sec) to 250 km periapsis alt. by 1 sol orbit.*
	[56+58]	Asc boiloff	157	30 day surf stay; calc: Boeing 'CRYSTORE' program
	[52]	Asc RCS prop	172	N2O4/MMH prop, Isp=280 sec, *Asc RCS dV =35 (m/sec)*
	Sum	***Total Asc propellant load***	15811	
	[63]	***Asc veh total mass***	22462	all masses in kg

synthesis model run#: marslander.dat;114

Mac chart: M Ref NTR MEV asc veh wt *STCAEM/bbd/22May90*

Ascent Cab - for Reference MEV Vehicle

Crew of 4, 3 day occupancy time Revision 2 5/22/90

	Element	*mass (kg)*	*Rationale*
Cab ECLSS	Atmospheric Revitization Sys/ Trace contaminant control assembly	123	CO_2 adsorption unit, expendable LiOH cartridge. Pre & postsorbent beds,catalytic oxidizer for particulate & contaminant control
	Atmosphere Control System	62	Total & partial press control; valves,lines & resupply/ makeup O_2 & N_2 and tanks
	Atmos. Composition & Monitor Assem.	55	O_2 & N_2 monitor for ACS, particulate & contaminant monitor for ARS
	Thermal Control Sys	40	Temp control: sensible liq. heat exchanger, ext radiator wt included in 'secondary structure' mass
	Temp. & Humidity Control	240	Condensing heat exchanger, fans, ducting
	Water Recovery and Management	45	Stored Potable water only
	Fire Detection & Suppression Sys.	113	Automatic sys w manual extinquishers as backup
	Waste Management Sys and Storage	-	Considered part of 'Man Systems'
	Asc cab ECLSS mass	**678**	Apollo style open ECLSS system
Cab Structure	Primary/Secondary Structure	519	Overpressurized (20 psia) on launch for structural integrity. Stiffening rings added at cylinder/endcap interface for added strength. Skylab derived triangular grid floor with beam supports on 6" centers. Support ring interface on pressure vessel to carry loads imposed by the floor and equipment during launch to aerocapture.
	Berthing ring/mechanism (1)	139	
	Berthing interface plate (1)	90	
	Windows	90	
	Couches	80	
	Hatches (2)	80	
	Asc cab Structure mass	**998**	

synthesis model run number: marsntr.dat.
Mac chart:M Ref MEVasc cab wt-ratio

Crew habitat module - MTV for 2016 NERVA NTR Ref Vehicle

Zero-g, Crew of 4, 434 day total trip time *Revision 6 5/22/90*

		Element	*mass (kg)*	*Rationale*
Crew Hab sys	[360]	Structure	8351	Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties
	[363]	ECLSS	4256	SSF derived with same degree of closure, sized for crew of 4 for 565 days
		Command/Control/Power		
	[364]	• Internal	1159	ECWS, DMS, batteries, other avionics/computing/monitoring eq, conditioning equip
	[i281]	• External Power	1539	Solar array,boom, power distribution,power management,fuel cell system
	[368-316]	Man systems	4121	Wts -all sys:SSFderived(as a funct. of crew size&occupancy time)for Mars missions
	[316]	Crew & effects	440	110 kg per person including personal belongings
	[373]	Spares/Tools	1496	Subsys component level spares. Life crit sys are 2 fault tolerent (approach of SSF)
	[247]	Radiation shelter	1802	Provides 10 g/cm2 protection + 3-5 g/cm2 provided by vehicle structure and equip
	[377]	Weight growth	2973	15% weight growth for dry mass excluding crew & effects and radiation shelter
	[237]	Airlocks	1530	2 x 765 kg external airlocks (shuttle type airlocks modified for MTV mission)
	[330]	EVA suits	0	EVA suits weight counted in MEV ascent cab weight statement
	[i60]	TTNC & GN&C platforms wt	863	3 platforms
	[378]	***MTV 'dry' crew hab mod wt***	28531	'dry' hab module represents structure and support systems equip & hardware that are dependant on crew size and independant of mission duration
	[371]	*On board equip resupply	986	Based on adjusted SSF resupply reqts for pot w, hyg w, ARS,TCS/THC &WMS
	[398]	*Consumables	4422	Crew of 4 for 434 days; food:2.04 kg/man/day, food pkg:0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man. food vol: 0.0055 m3/man/day, other: 0.0018.
	Sum	***MTV crew mod 'wet' wt***	5408	
Crew mod supp. sys	[i65]	*Transfer science equipment	1000	Inb and outb MTV science hardware and supplies
	[227]	Art-g RCS spin up propel	0	zero g environment
	[i61+126	Art-g tether mass	0	zero g environment
	[i57]	Remote Manipulator-arm Sys	0	all large external self assembly hardware left in LEO
	[734]	***Hab mod support sys wt***	1000	
	124-230	***MTV crew mod & support systems weight***	34939	This wt refects the Boeing ref crew of 4 mod downloaded by 1370 kg of consumables and 318 kg of onboard resupply because the shorter 2016 opposition mission (434 days vs 565 chem/AB ref). The mod 'dry' wt represents a SSF type closed ECLS Sys (air >99%, water >95%) that serves the crew with 2 fault tolerence on all life critical sys except structure. Its wt varies primarly with crew size. consumables wt varies with crew size and mission duration.

* *MTV hab mod consumables, resupply, and transit science dependant on mission duration, and free abort requirement. i.e. crew mod 'wet' wt will vary for different missions*

Mac chart: M Ref MTV mod wt-rationale
synthesis model run# marschemmtv.dat;21

2016 NERVA NTR Reference Vehicle:

Frame, propulsion sys, & shield wt: Isp = 925 *Rev 5 5/22/90*

[159]	Spacecraft frame (truss) struct	3000	Truss struct: Graphite epoxy, Ec= 16 Msi, Den=0.06 lb/in3, 2 m by 35 m SSF type
[183]	RCS inert wt	800	estimate scaled from RCS propellant load
[709]	Main prop line wt	451	Main line from tank lines to reactor, L=35m,d wall s steel;dens=7833kg/m3,t=0.8mm
[160]	Mass growth	638	15% mass growth
[518]	Engines wt (1)	9684	***75k lbf Thrust***, wt estimate: NASA/LeRC propul task order (Westinghouse, others)
[543]	Engine shield wt (1)	4500	4500 lbf shadow shield wt from LeRC propul task order
[118]	RCS prop wt	704	Transfer RCS dV = 20, Isp = 300, storable biprop
[696]	***Frame & propul 'dry' wt***	19777	

Mars dep & Earth capt stages - 2016 NERVA NTR Ref Veh

14% tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 sec Rev 5 5/22/90

		Element	Mass (km)	Rationale
Mars dep	[128]	Mars dep usable prop load	53168	***Mars dep dV= 3900 m/s; eng Isp=925 sec,*** H2 density =70.8
	[703]	Mars dep prop residuals	1063	2% residuals/reserve left after boiloff,burn and cooldown
	[699]	Mars dep burn 'cooldown' prop	1595	3% post burn prop for reactor cooldown; no thrust/Isp counted in this approximation
	[498]	Mars dep stg outbound boiloff	1792	Out b boiloff for given MLI & VCS insul.;no refrig,based on Boeing 'CRYSTORE'
	[545]	Mars dep stg inorbit boiloff	361	31.5 day inorbit stay time
	[122]	Inbound midcourse prop	1266	Inb midc maneuver dV=90 m/s; done by main propulsion system
	[711]	**Tot Mars dep stg prop load**	**59245**	total at time of TMI burn
Earth capt	[561]	Earth arr stg usable prop tot	23638	***Earth arr dV=2629 m/s; propulsive burn capture into 500 km by 24 hr ellip orbit***
	[704]	Earth arr stg prop residuals	472	2% residuals/reserve left after boiloff,burn and cooldown
	[700]	Earth arr stg 'cooldown' prop	709	3% post burn prop for reactor cooldown; no thrust/isp counted in this approximation
	[562]	Earth arr stg outbound boiloff	2937	434 day b.off period; additional b.off from this tank also accounted in M dep p b.off
	[563]	**Total Earth arr stg prop load**	**27756**	Total at time of TMI burn
	[570]	**Total combined prop load**	**87001**	M dep/E arr prop: put in 1 tank along veh centerline aids NTR radiation attenuation
Common tank				***1 continuous reinforced Silicon Carbide/Al metal martrix tank:*** dia:10.m, L:19.0m,
	[683]	Single M dep/E arr tank wt	5986	filament wound; dens= 2436 kg/m3; 37ksi Wk. stress; tank skin thickness = 4.0 mm
	[684]	MLI wt	1978	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x dens
	[685]	Vapor cooled shield wt	1563	2 VCS - 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[686]	Meteoriod shield wt	1323	One 0.80 mm sheet of Al; comparsion: SSF plans 0.8 mm, Mariner 9 used 0.4 mm
	[687]	Propel line/valves wt	225	length =10 m,double wall stainless steel H2 prop line; density= 7833 kg/m3, t=0.8mm
	[689]	Mass growth wt	2770	25% wt growth for tank shell, MLI,VCS,meteor shield,prop line & attachment
	[565]	Sum of inerts:single tank	13845	Total for single tank with all tank related inerts.
	[566]	Total for 1 tank	13845	Overall tank fraction [571] = 13.7 %
	[572]	***Combined Mars dep/Earth arr tank set & propellant load***	**100846**	Total for ' Mars dep/Earth arr tank set ' at time of TMI burn
	[171]	***IMLEO***	**735190**	

Mac Chart: M NTR E arr wr

Boeing vehicle synthesis model run #: marsntrmtv.dat;55

Earth dep & Mars capt stages - 2016 NERVA NTR Ref Veh

14 % tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 *Rev 5 5/22/90*

	[586]	Earth dep usable propel tot	272520	***Earth dep dV: 4182 m/s (includes 200 m/s gloss for 2 burn E dep) ; Isp = 925 sec***
	[701]	Earth dep prop residuals	5450	2% residuals/reserve left after boiloff, burn prop, and cooldown
	[697]	Earth dep burn 'cooldown' prop	<u>8176</u>	3% post burn prop for reactor cooldown: no thrust/Isp counted for this estimated %
	[705]	**Tot Earth dep stg prop load**	**286146**	
Earth dep stg wt				***2 continuous reinforced Silicon Carbide /Al metal matrix tanks:*** dia: 10.0 m,
	[668]	Single tank wt (cyl/ellip ends)	9198	L:30.0m, dens= 2436 kg/m3; 37ksi wk. stress, thickness = 4.0 mm, root 2 ellip ends
	[669]	MLI wt	3040	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x density
	[670]	Vapor Cooled Shield wt	2401	2 VCS: at 2 x 0.13mm Al outer sheets with 0.57 kg/m2 honeycomb core each
	[671]	Meteoriod shield wt	2039	One 0.80 mm sheet of Al; comparison: SSF uses 0.8 mm, Mariner 9 used 0.40 mm
	[i312]	Tank/frame attachment	400	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[672]	Tank feed prop line wt	159	Short prop line from tank to main prop line:double wall, stainless steel: 10 meter
	[674]	<u>Mass growth wt</u>	<u>4309</u>	25% wt growth for tank inert,MLI,VCS,meteor shield,prop lines, tank/veh attachment
	[588]	Sum of single tank inerts	21546	Total single tank inert wt;
	[592]	Total for 2 tanks	<u>43092</u>	Total for ' Earth dep tank set ': inert wt; Overall tank fraction [593] = 13.1 %
	[597]	***Earth Dep stage tot wt***	**329238**	Total Earth dep stg weight at time of Trans Mars Injection burn
	[610]	Mars arr usable prop tot	137450	***Mars arr dV: 3870 m/s; eng Isp=925,*** H2 density = 70.8
	[702]	Mars arr prop residuals	2749	2% residuals /reserveleft after boiloff, burn prop, and cooldown
	[698]	Mars arr burn 'cooldown' prop	4124	3% post burn prop used for reactor cooldown; prelim;based on Westingh. estimate
	[611]	Mars arr stg outbound boiloff	3349	Boiloff for given MLI,VCS and Outb trip time; based on Boeing's 'CRYSTORE' prog
	[121]	<u>Outbound midcourse prop</u>	<u>4008</u>	Outb midc maneuver dV = 120 m/s; done by main propulsion from M arr tanks
Mars arr stg wt	[612]	**Tot Mars arr stg prop load**	**151680**	
	[633]	Single M arr tank wt	5346	***2 SiC/Al metal matrix tanks;*** dia:10.0 m, L:17.0 m, dens=2436 kg/m3;thick=4.0mm
	[634]	MLI wt	1737	MLI: density = 32 (kg/m3); 200 layers (4 inches) at 20 layers/cm
	[635]	Vapor cooled shield wt	1396	2 VCS: at 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[636]	Meteoriod shield wt	1185	One 0.80 mm sheet of LiAl: assumption-LEO assembley in protective hanger
	[i312]	Tank/frame attachment	400	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[637]	Tank feed prop line wt	159	Double wall, stainless steel 10 meter H2 propellant line;dens= 7833 kg/m3, t=0.8mm
	[639]	<u>Mass growth wt</u>	<u>2563</u>	25% wt growth for tank inert,MLI,VCS,meteor shield,prop line&tank/veh attachment
	[614]	Sum of single tank inerts	12786	Total for single tank, with all inerts.
	[615]	Total for 2 tanks	<u>25572</u>	Overall tank fraction [620] = 14.5 %
	[617]	***Mars Arr stage wt***	**177252**	wt at time of Earth departure

4

Artificial Gravity Option

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Reference

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Nuclear Thermal Rocket Vehicle Artificial Gravity Configuration

The NTR artificial gravity configuration spins nominally at 3.98 rpm outbound (56.5 m to create 1g) and 3.83 rpm inbound (61 m to create 1g). The truss used is similar to the 0g configuration but optimized for a gravity field, thus increasing mass. The spin radius of the vehicle does not change very much because the Earth departure and Mars arrival tanks are placed on the vehicle CM. The vehicle has nominally 4 spin-up/spin-down cycles and used the Earth arrival propellant and reactor as countermass.

The NTR configuration is probably the least affected by artificial gravity of any of the reference vehicles. The main changes to the vehicle are a longer, heavier truss to facilitate gravity, and added RCS and TMI/TEI propellant. Other complications of artificial gravity are the spin-up/spin-down cycles and the "despun joints" required for power and communication.

Artificial Gravity (g_a) Assessment
Assumptions

A 1g gravity level was assumed for this study over partial g because the minimum gravity level required to offset physiological deterioration is not known. The rotation rate was set to be no more than 4 rpm, which is based on experimental data in the Pensacola Slow Rotation Room (1960's) on human adaptation. The crew compartments are contiguously pressurized during all mission phases, and the crew modules are to be oriented with the long axis parallel to the spin vector to offset the Coriolis effect along major circulation paths. Connections between habitation and the countermass are either tethers or a truss rather than a pressurized tunnel because, since all crew compartments are contiguous, the is no need for an IVA transfer.

Artificial Gravity (g_a) Assessment

Assumptions	Rationale
1g gravity level	• Earth-normal conditioning for exploration in surface EMU
Rotation rate ≤ 4 rpm (56 m)	• Generally accepted range for vestibular disturbance tolerance
Contiguous crew compartments	• Maximize available volume • In-flight simulation and training • Contingency operations
Truss and tether connections • Tethers are "ribbon" shaped	• Avoids mass penalty • Not needed for contiguous volumes • Facilitates conductors
Module orientation parallel to spin vector	• g level consistency; minimizing vestibular disturbance • Mass properties quasi-isotropic to first order

/STCAEM/sdc/12June90

Artificial Gravity (g_a) Assessment Assumptions

- Gravity level
  - 1g chosen over partial g (less than 1g)
- Rotation rate
  - $\leq$ 4 rpm (4 rpm at 56 m nominally)
- Crew compartments
  - contiguously pressurized throughout all mission phases
- Connection
  - truss and tethers rather than a pressurized tunnel
  - multiple tethers are used that are "ribbon" shaped in cross section
- Module orientation
  - long axis parallel to spin vector

g_a NTR Vehicle Features

- Nominal spin rate
  - 3.98 rpm outbound (56.5 m to create 1g)
  - 3.83 rpm inbound (61 m to create 1g)
- 7 m square cross-section truss
- Earth departure, Mars arrival, and Mars departure tanks placed on CM
  - minimize CM movement when tanks are dropped
- Nominally 4 spin-up/spin-down cycles
  - Earth arrival propellant and engine used as countermass

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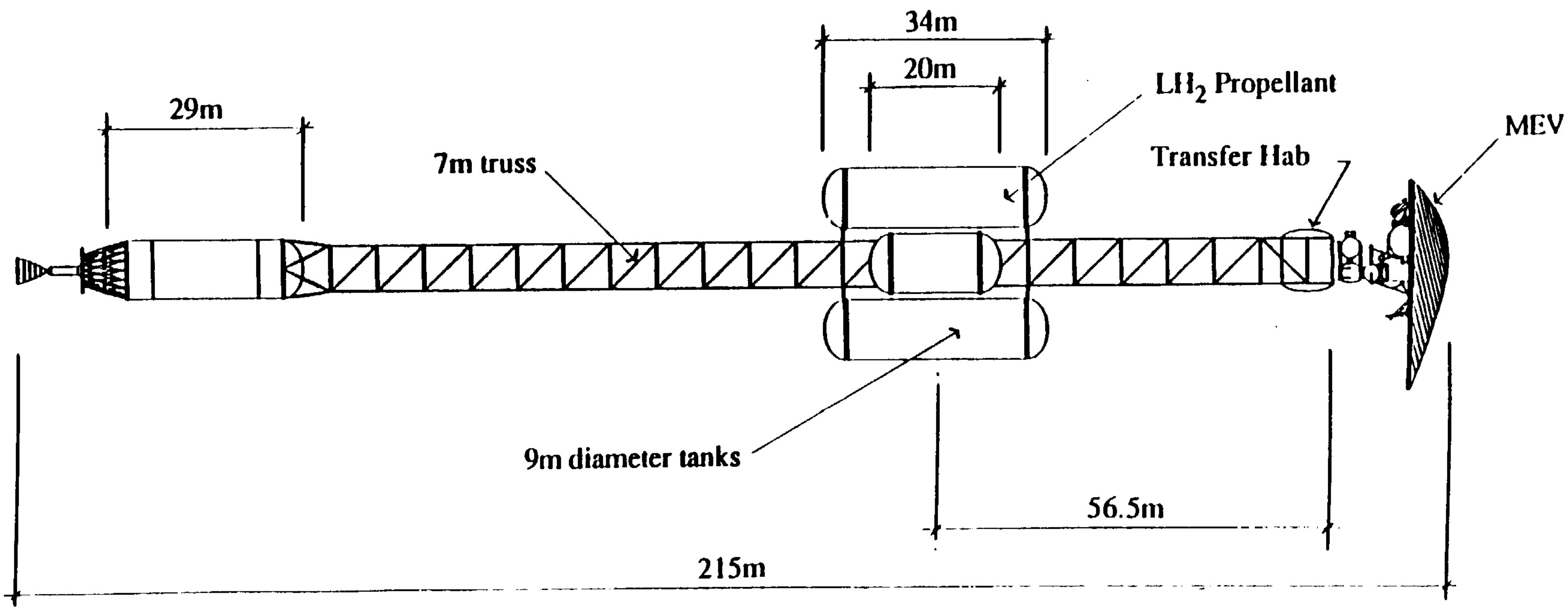
g_a NTR Configuration

The NTR artificial gravity configuration makes use of the Earth arrival propellant as a countermass and has a rigid truss as connection. The Earth departure, Mars arrival, and Mars departure tanks are located at the CM so, as dropped, minimally disturb the total vehicle CM (4.5m total movement). The Earth arrival propellant is also used as radiation protection for the crew areas from the NERVA type engine.

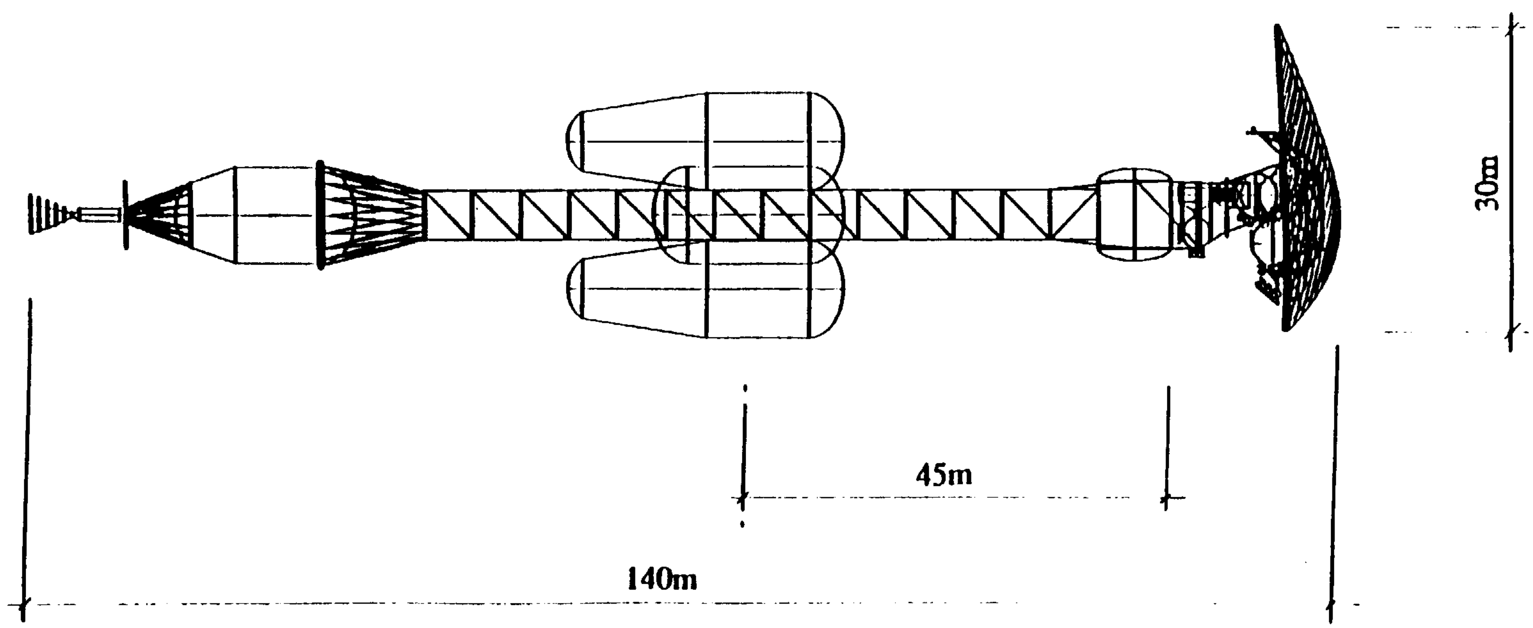
g_a NTR Configuration



MTV	34.9
MEV	73.1
Prop./Structure	679.3
IMLEO	**787.3 t**

/STCAEM/sdc/31May90

g_a (1/3g) NTR Configuration



MTV	34.9
MEV	73.1
Prop./Structure	643.1
IMLEO	751.1 t

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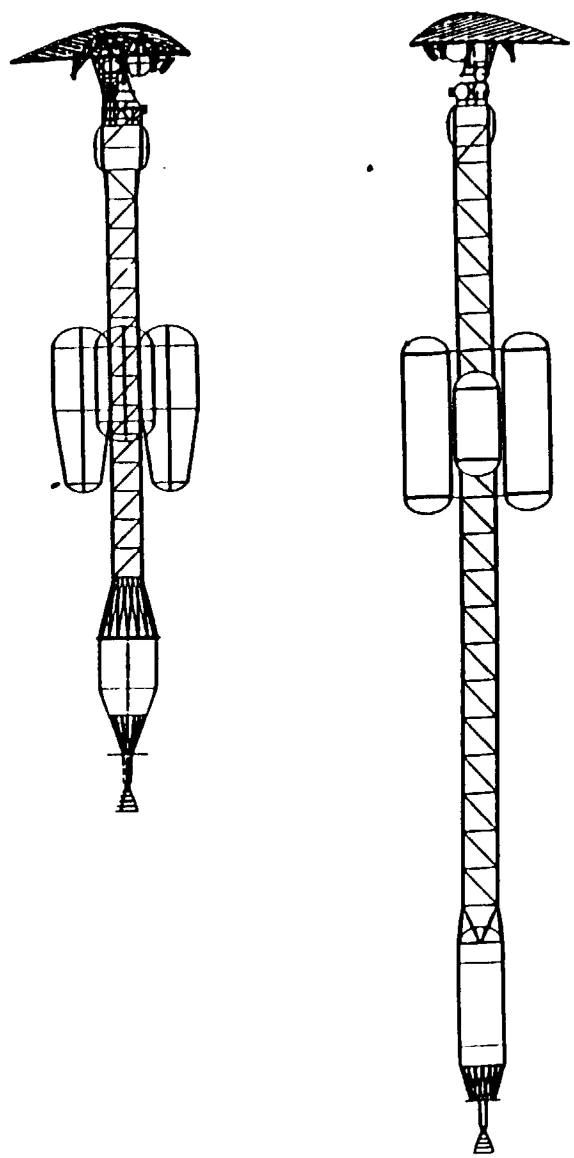
g_a NTR Mass Statement



Element	Mars-g	Earth-g
MEV desc aerobrake	7000	7000
MEV ascent stage	22464	22464
MEV descent stage	18659	18659
MEV surface cargo	25000	25000
MEV total	73118	73118
MTV crew hab module 'dry'	28531	28531
MTV consummables & resupply	5408	5408
MTV science	1000	1000
MTV crew hab sys tot	34939	34939
MTV frame, propulsion, & shield wt	20033	23031
*MTV added Art-g RCS hardware	650	650
*MTV added Art-g RCS prop	2628	7010
Earth Orbit Capture (EOC) prop	28305	29714
Trans Earth Inject (TEI) prop	60829	64541
EOC/TEI common tank wt	14129	14789
Mars Orbit Capture (MOC) prop	154250	160580
MOC tanks	25967	26788
Trans Mars Inject (TMI) prop	292320	306410
TMI tanks	43891	45719
ECCV	0	0
Cargo to Mars orbit only	0	0
IMLEO	751077	787289

* Art-g spinup sys wt, in addition to nominal maneuver RCS sys

all masses in kg's

Mac chart: M Art-g 2016 NTR cover pg
synthesis modelrun#:marsnrmtv.dat;180,181

/STCAEM/sdc/08June90

g_a NTR Penalty Assessment

This chart outlines the penalties of using artificial gravity with the NTR configuration. The penalties for this concept are much less than the Cryo/AB version. There is a limited mass increase because the only major difference is the longer truss and despun joints, which subsequently increases propellant loads.

g_a NTR Penalty Assessment

- 7 % added mass
  - Heavier truss (vehicle "hung" from center of rotation)
  - Longer truss
  - Added RCS and propellant
  - Added TMI/TEI propellant
- Spin-up/spin-down cycles
  - Mid-course correction problems
- "De-spun" joints for power and communication

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Options / Alternatives

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Artificial Gravity Options/ Alternatives

An investigation was made of using an artificial gravity NERVA-type NTR vehicle on a low energy 2025 conjunction mission to carry two and three "mini-landers" (small MEV's). These would be used for short duration manned surface missions with some cargo capability at more than one landing site or as rescue backups for a damaged lander. They were traded for IMLEO varying crew size with Isp and IMLEO to cargo weight as well as the level of the gravity conditions in transit. Shown are the configurations and weight statements developed for the two and three lander artificial gravity vehicles for a 2005 conjunction mission with 0.3 Earth gravity (Mars gravity level).

2005 conjunction mission NTR vehicle

A NERVA NTR vehicle design was done for a low energy 2005 conjunction mission which is characterized by a 983 day trip time with 482 days at Mars. For this mission a ECCV was taken for crew return to Earth and the vehicle was expended. The engine for this vehicle was the reference NERVA derivative of 9684 kg, Isp of 925 sec and a shadow shield wt of 4.5 t. A nominal crew of 6 was taken. Both MOC propellant and TEI propellant were carried in a single aft tank since the dV for these burns were quite small when compared to the higher dV of opposition trajectories. The payload carried to Mars was a follows:

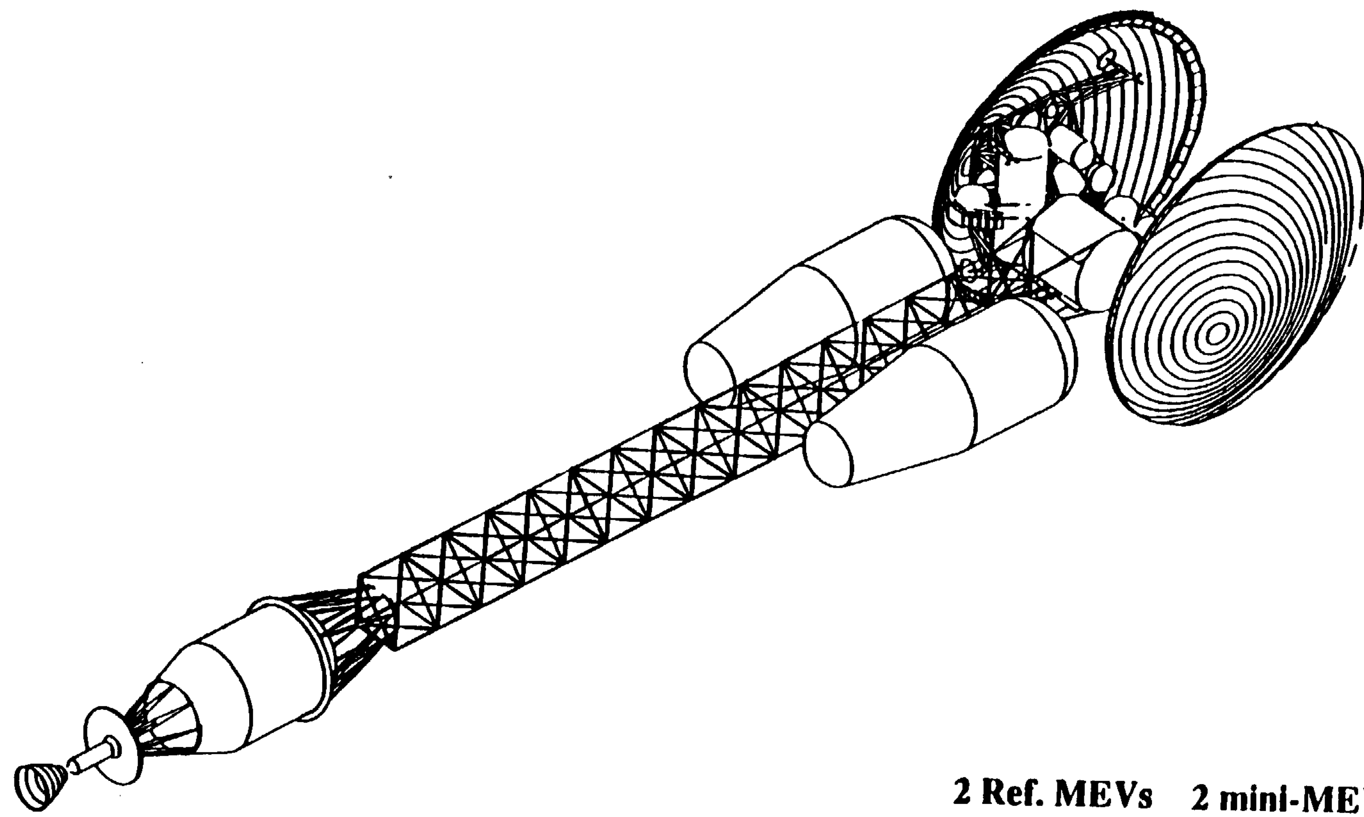
(1) 2 reference MEVs and 30 t of cargo to Mars orbit
(2) 2 mini MEVs and 30 t of cargo to Mars orbit
(3) 2 mini MEVs and 10 t for cargo to Mars orbit

For case (1) the following sensitivities were evaluated:

(1) NTR Isp on IMLEO
(2) Crew size affect on IMLEO
(3) Cargo to Mars orbit affect on IMLEO
(4) Vehicle expended vs vehicle recovered modes

Results are shown on the following two charts

g_a (1/3g) NTR Configuration
(2005 Conjunction Mission)



	2 Ref. MEVs 30 t Cargo	2 mini-MEVs 30 t Cargo	2 mini-MEVs 10t Cargo
MEV	146.2	79.5	79.5
Transfer Hab	62.2	62.2	62.2
Prop./Cargo/Str.	396.9	323.5	281.1
IMLEO	605.3 t	465.2 t	422.8 t

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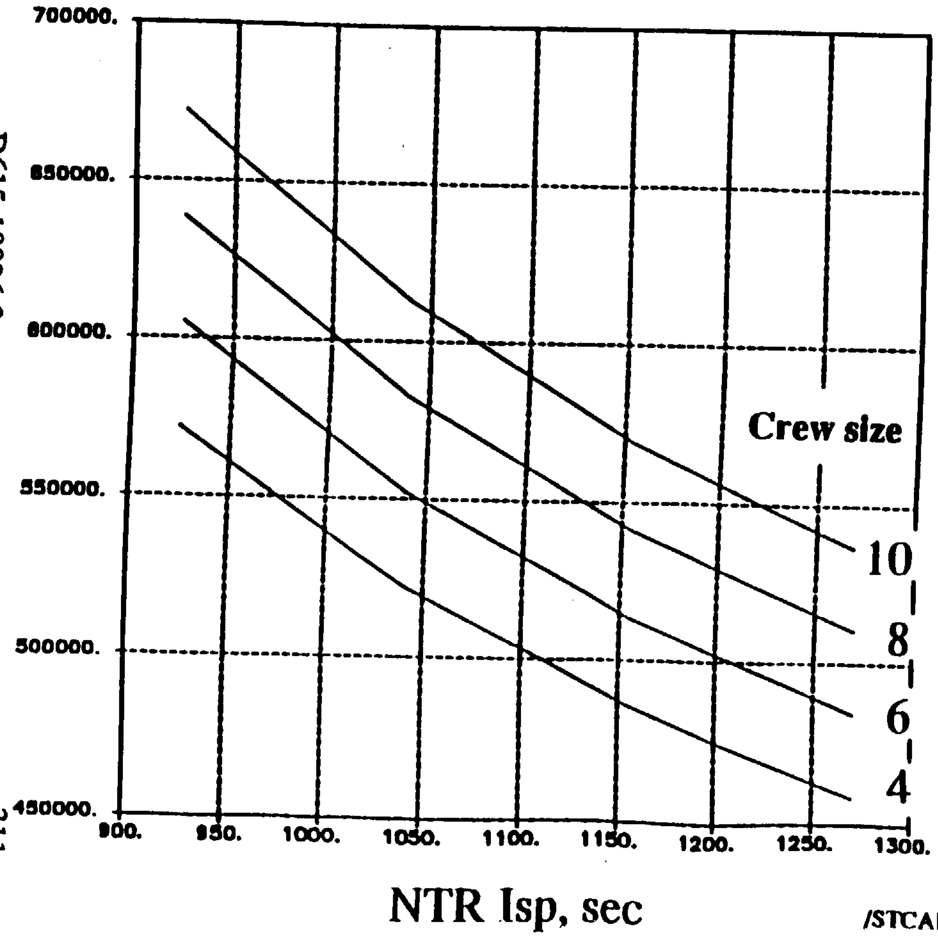
2005 Conjunction NERVA NTR Vehicle Trades

Effect of Crew Size, Eng Isp, Mars Orbit Cargo, & Earth Return Option on Veh IMLEO

983 day trip time, 2 x 73 t MEV's, Art-g (Mars-g) configuration, one 9684 kg 75000 lbf eng, 4.5 t sh....
Nominal case: crew of 6, Isp=925, 30k cargo to orbit, ECCV return, IMLEO=605345 (kg)

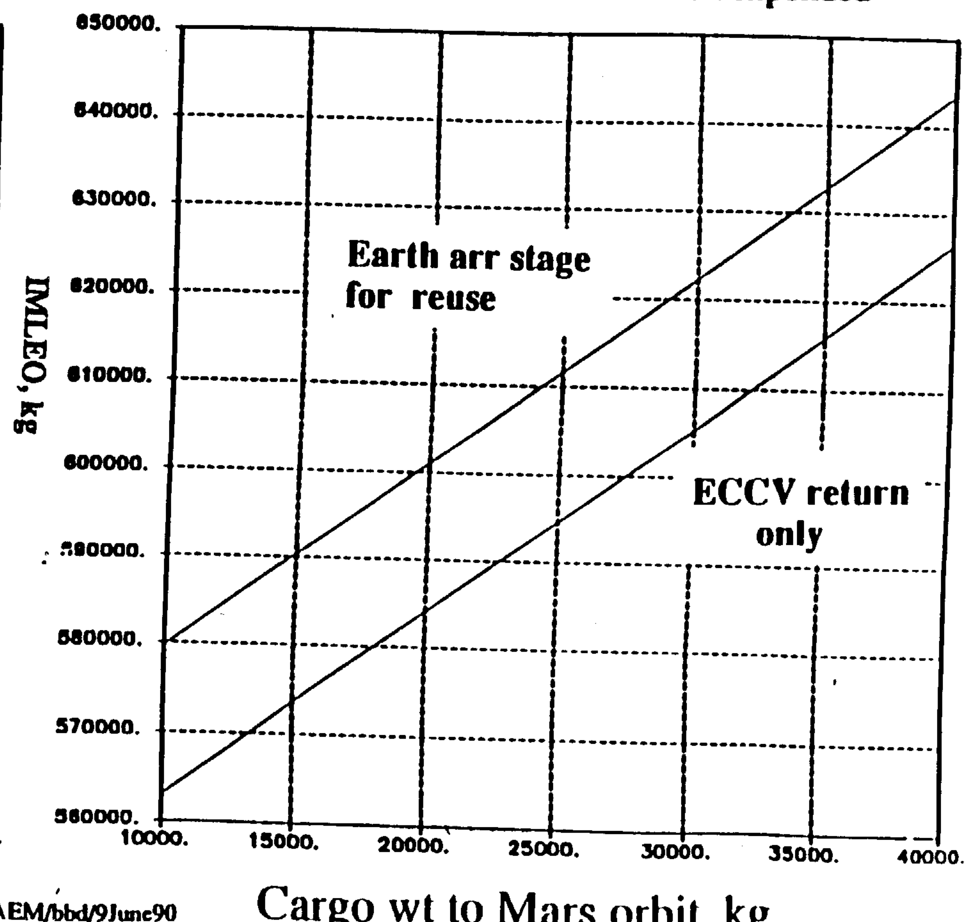
IMLEO vs NTR eng Isp and crew size

- 4 t IMLEO savings per every 10 sec Isp increase
- 17 t IMLEO increase per single crew member addition above 4



IMLEO vs cargo to orbit and Earth return option

- 22 t IMLEO increase for every 10 t delivered to Mars orbit
- 16 t IMLEO increase to recapture veh in 500 km by 24 hr Earth orbit for reuse vs ECCV return w veh expended



/STCAEM/bbd/9June90

g_a Mass Summary

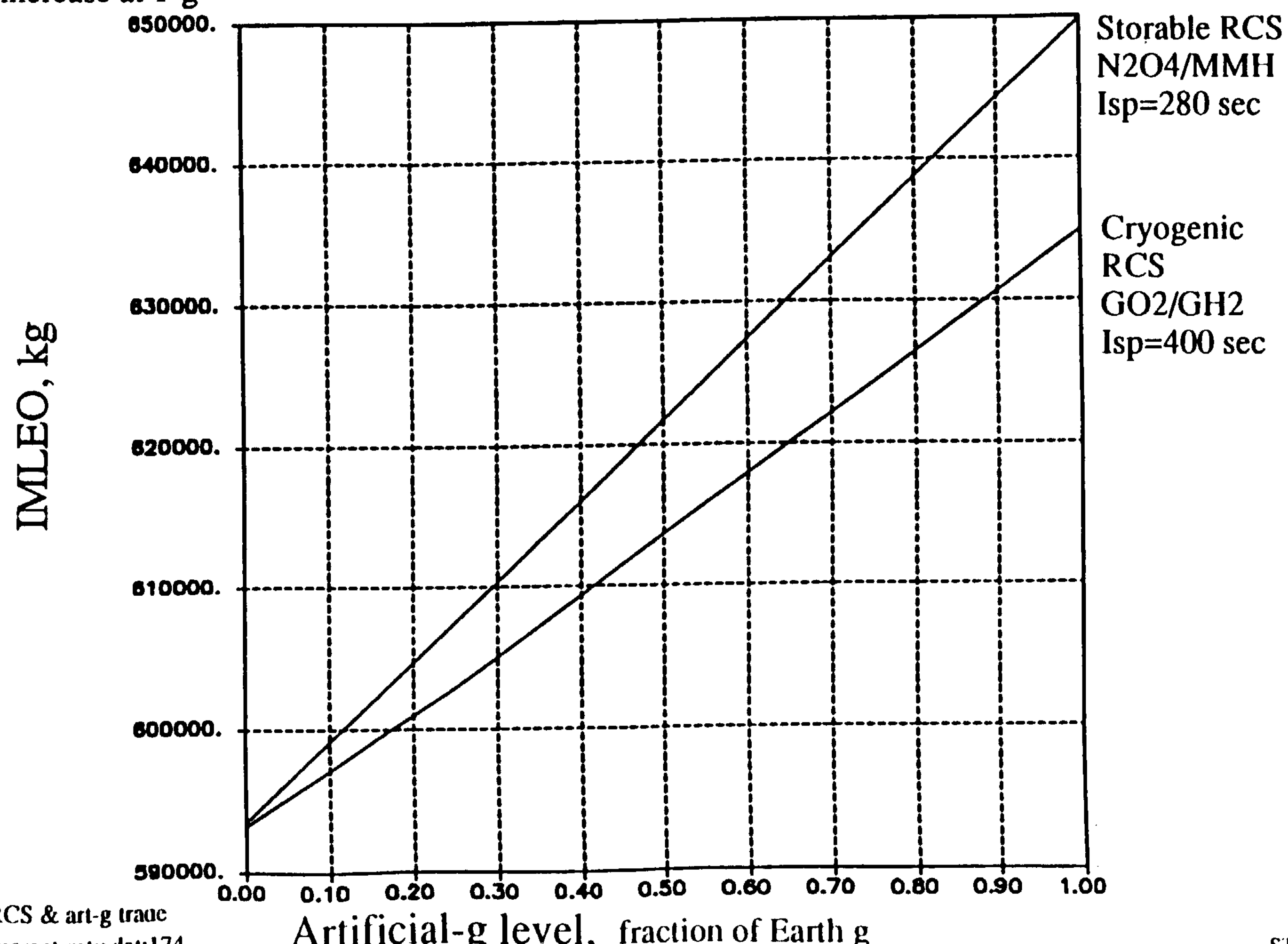
This chart shows the relative mass of the Cryo/AB and the NTR artificial gravity configurations as compared to all the reference 0g configurations. The Cryo/AB configuration trades very poorly in artificial gravity, whereas the NTR configuration has only minor mass impact.

2005 Conjunction NERVA NTR Vehicle Art-g Trade

Effect of Artificial-g level & RCS propellant choice on Vehicle IMLEO

983 day trip time, 2 x 73 t MEV's, Art-g (Mars-g) configuration, one 9684 kg 75000 lbf eng, 4.5 t sh ECCV return only, no veh reuse. Nominal case: crew of 6, Isp=925, 30k cargo to orbit, IMLEO=605345 (kg)

RESULTS:
- 25 t IMLEO increase of 1-g over 1/3 g (Mars-g)
- 6 t IMLEO increase for storable biprop RCS (Isp=280 s) over GO2/GH2 RCS (Isp=400 s) at Mars-g
- 15 t increase at 1-g



Mac chart: 2005 conj NTR RCS & art-g trade
Veh synthesis model run #: marsntrmtv.dat;174

STCAEM/bbd/26June9?

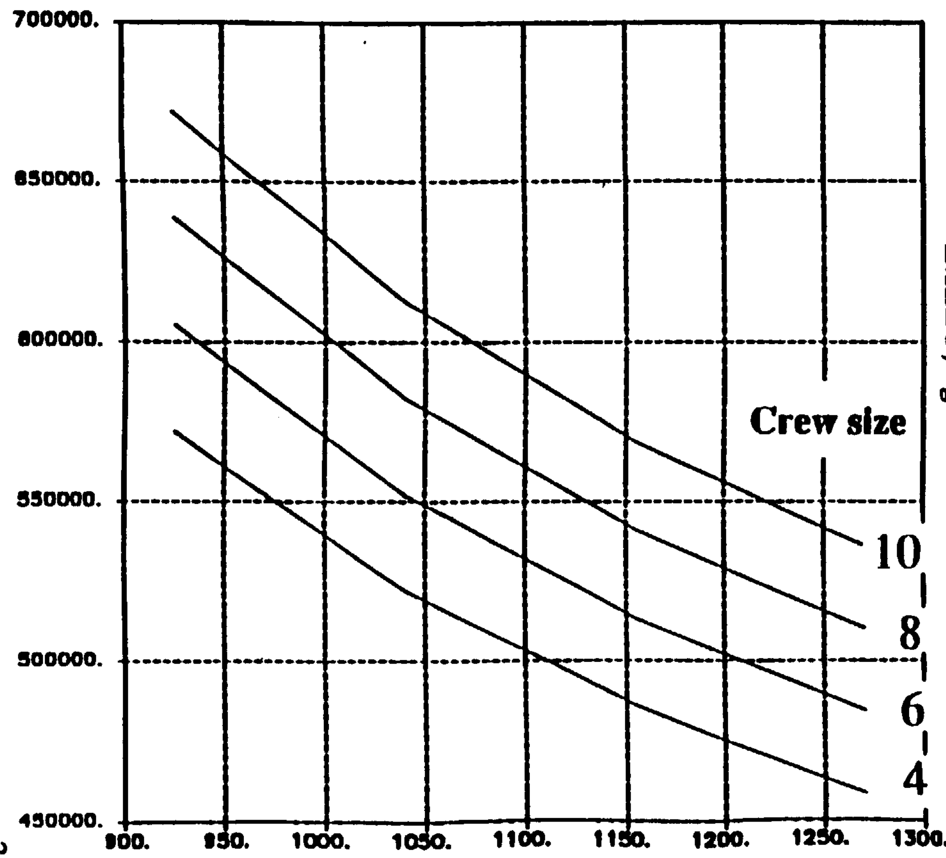
2005 Conjunction NERVA NTR Vehicle Trades

Effect of Crew Size, Eng Isp, Mars Orbit Cargo, & Earth Return Option on Veh IMLEO

983 day trip time, 2 x 73 t MEV's, Art-g (Mars-g) configuration, one 9684 kg 75000 lbf eng, 4.5 t sh...
Nominal case: crew of 6, Isp=925, 30k cargo to orbit, ECCV return, IMLEO=605345 (kg)

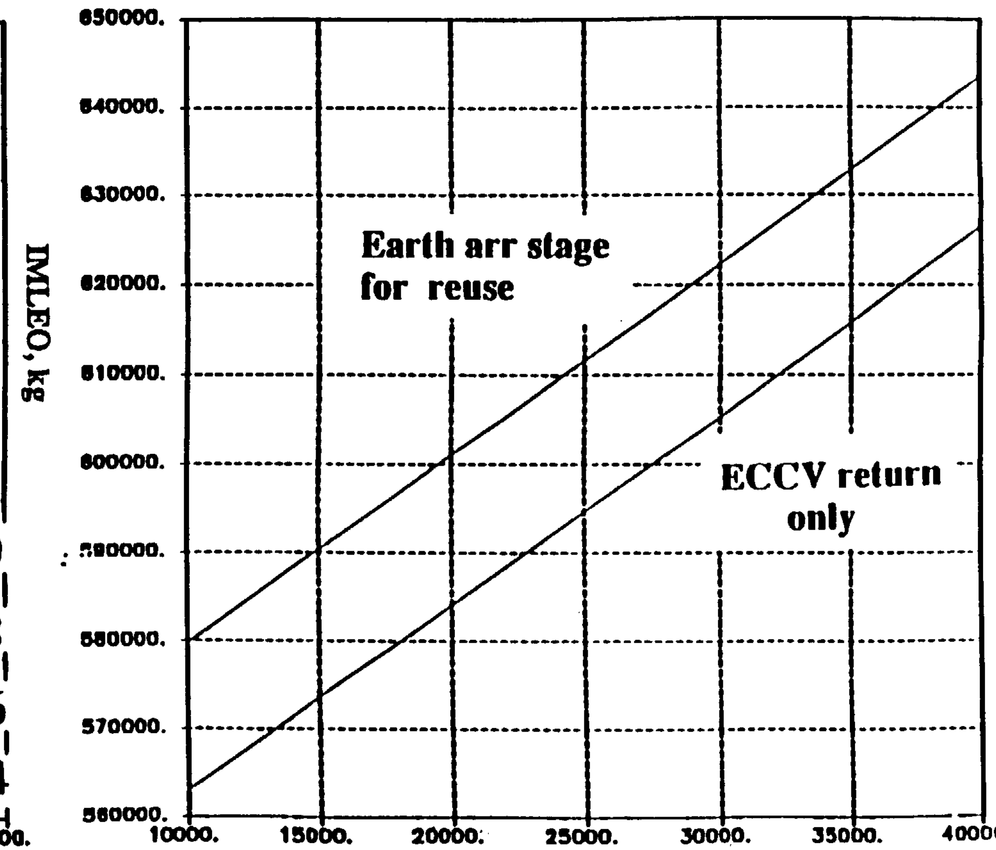
IMLEO vs NTR eng Isp and crew size

- 4 t IMLEO savings per every 10 sec Isp increase
- 17 t IMLEO increase per single crew member addition above 4



IMLEO vs cargo to orbit and Earth return option

- 22 t IMLEO increase for every 10 t delivered to Mars orbit
- 16 t IMLEO increase to recapture veh in 500 km by 24 hr Earth orbit for reuse vs ECCV return w veh expended



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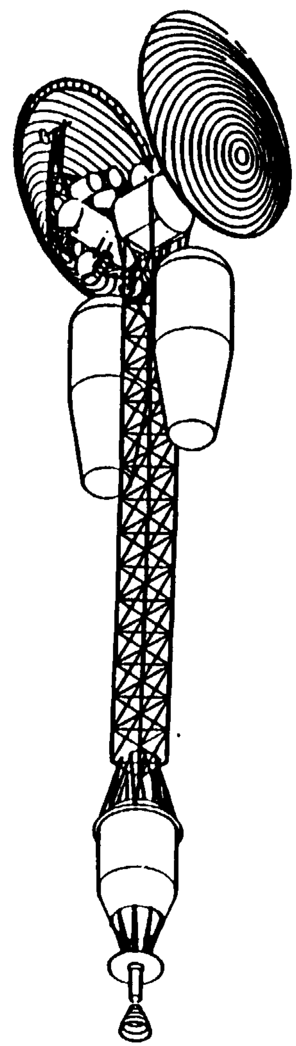
Art-g (Mars-g) NTRVehicle for 2005 Conjuction Mission

Nominal payload: 2 x 73t MEV's (desc only aeroshell), 30k science cargo to mars orbit only, ECCV crew return, no vehicle reuse, Crew of 6, 983 day trip time

TMI dV: 4267 m/s (includes 300 m/s g-loss), MOC dV: 863 m/sec, TEI dV: 1179 m/s 6/7/90



Element	*2 Ref MEVs 30k cargo*	*2 mini MEV's 30k cargo*	*2 mini MEV's 10k cargo*
MEV desc only aerobrake	7000		
MEV ascent stage	22464		
MEV descent stage	18659		
MEV surface cargo	25000		
MEV total	73118	39752	39752
x 2	146236	79504	79504
MTV crew hab module 'dry'	34775	34775	34775
MTV consumables & resupply	19398	19398	19398
MTV crew habit module total	54173	54173	54173
MTV frame,propulsion & shield wt	20033	20027	20027
*MTV Art-g added RCS hardware	650	650	650
*MTV Art-g added RCS prop	6348	5134	4588
Trans Earth Injec (TEI) prop	19648	19185	19042
Mars Orbit Capture (MOC) prop	35166	26955	24470
MOC/TEI/EOC common tank wt	9669	8543	8202
MTV propulsion/frame/prop total	91514	80494	76979
TMI tanks wt	36911	29732	27563
Trans Mars Injec (TMI) prop	238510	183220	166520
TMI stage total	275420	212950	194083
ECCV	8000	8000	8000
MTV Cargo to Mars orbit only	30000	30000	10000
IMLEO	605343	465121	422739

* Art-g spinup wt additions

Mac chart: M NTR 2005 Conj cover pg
synthesis model run# marschemmtv.dat;177,178,179

./STCAEM/bbd/31May90

Crew hab mod - MTV for 2005 Conj Art-g (Mars-g) NTR Veh

Art-g [Mars g], Crew of 6, 983 day total trip time 6/5/90

Section	Ref	Element	mass (kg)	Rationale
Crew hab nodule dry weight	[360]	Structure	10878	Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties
	[363]	ECLSS	5743	SSF derived with same degree of closure. supports a crew of 6 for 983 days
		Command/Control/Power		
	[364]	• Internal	1159	ECWS, DMS, batteries, other avionics/computing/monitoring eq, conditioning equip
	[i281]	• External Power	1539	Solar array,boom, power distribution,power management,fuel cell system
	[368-316]	Man systems	4457	Wts -all sys:SSFderived(as a funct. of crew size&occupancy time)for Mars missions
	[316]	Crew & effects	660	110 kg per person including personal belongings
	[373]	Spares/Tools	1803	Subsys component level spares. Life crit sys are 2 fault tolerent (approach of SSF)
	[247]	Radiation shelter	2439	Provides 10 g/cm2 protection + 3-5 g/cm2 provided by vehicle structure and equip
	[377]	Weight growth	3705	15% weight growth for dry mass excluding crew & effects and radiation shelter
	[237]	Airlocks	1530	2 x 765 kg external airlocks (shuttle type airlocks modified for MTV mission)
	[330]	EVA suits	0	EVA suits weight counted in MEV ascent cab weight statement
	[i60]	TTNC & GN&C platforms wt	862	3 platforms
	[378]	*MTV 'dry' crew hab mod wt*	34775	'dry' hab module represents structure and support systems equip & hardware that are dependant on crew size and independant of mission duration
Consumables and resupply	[371]	*On board equip resupply	3605	Based on adjusted SSF resupply reqts for pot w, hyg w, ARS,TCS/THC &WMS
	[398]	*Consumables	15793	Crew of 6 for 983 days; food:2.04 kg/man/day, food pkg:0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man. food vol: 0.0055 m3/man/day, other: 0.0018.
	Sum	*MTV crew mod 'wet' wt*	19398	
	[i65]	*Transfer science equipment	0	Inb and outb MTV science hardware and supplies
	[i57]	Remote Manipulator-arm Sys	0	all large external self assembly hardware left in LEO
	124-230	*MTV crew mod & support systems weight*	54173	This wt refects the Boeing ref crew module subsystems modified for an additional 2 crew members and a much longer mission duration. The mod 'dry' wt represents a SSF type closed ECLS Sys (air >99%, water >95%) that serves the crew with 2 fault tolerence on all life critical sys except structure. Its wt varies primarly with crew size. consumables wt varies with crew size and mission duration.

* *MTV hab mod consumables, resupply, and transit science dependant on mission duration, and free abort requirement. i.e. crew mod 'wet' wt will vary for different missions*

Mac chart: M NTR 2005 Conj mod Art-g synthesis model run# marsntrmtv.dat;178

!STCAEM/bbd/31May90

2005 Conj Art-g (Mars-g) NERVA NTR Vehicle:

Frame, propulsion sys, Artificial-g sys, & shield wt: Isp = 925 *6/5/90*

	Element	mass (kg)	Rationale
[159]	Spacecraft frame (truss) struct and aft tank struts wt	3480	Truss struct: Graphite epoxy, Ec= 16 Msi, Den=0.06 lb/in3, 5 m by 70 m SSF type
[183]	Maneuver RCS inert	1300	Systems fore and aft that fire simultaneously to provide rotation about vch Cg
[709]	Main prop line wt art-g	451	Main line from tank lines to reactor, L=40 m,d wall s steel;dens=7833kg/m3,t=0.8mm
[160]	Mass growth	785	15% mass growth
[518]	Engines wt (1)	9684	*75k lbf Thrust*, wt estimate: Stan Borowski of NASA/LeRC
[543]	Engine shield wt (1)	4500	Shadow shield wt from NASA/LeRC propulsion task order
[118]	Maneuver RCS prop wt	483	MTV maneuver RCS dV = 20, Isp = 400, gaseous O2/H2
[696]	***Frame & propul 'dry' wt***	20683	
[758+759]	Outb 1st spinup/down prop	2184	Outbound leg, 4 RPM Mars-g spin untill spindown for outb midc correction burn
[756+757]	Outb 2nd spinup/down prop	2175	Outbound leg, from midc correction to Mars capture
[754+755]	Inorbit spinup/down prop	846	inorbit spinup
[752+753]	Inb 1st spinup/down prop	572	Inbound leg, 4 RPM Mars-g spin untill spindown for Inb midc correction burn
[750+751]	Inb 2nd spinup/down prop	570	Inbound leg, from midc correction to Earth capture
[761]	***Total Art-g spinup/down prop***	6347	Total RCS prop for all 5 Art-g spin maneuvers

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Mac Chart: M NTR 2005 Conj Frame Art g

Synthesis model run #: marsntrmtv.dat;178

Mars dep & Earth capt stages - 2005 Conj Art-g (Mars-g) NTR Veh

***14% tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 sec** 6/5/90*

		Element	Mass (km)	Rationale
Mars dep	[128]	Mars dep usable prop load	13220	***Mars dep dV= 1179 m/s; eng Isp=925 sec,*** H2 density =70.8
	[703]	Mars dep prop residuals	264	2% residuals/reserve left after boiloff,burn and cooldown
	[699]	Mars dep burn 'cooldown' prop	396	3% post burn prop for reactor cooldown; no thrust/Isp counted in this approximation
	[498]	Mars dep stg outbound boiloff	2293	Out b boiloff for given MLI & VCS insul.;no refrig,based on Boeing 'CRYSTORE'
	[545]	Mars dep stg inorbit boiloff	2239	482 day inorbit stay time
	[122]	Inbound midcourse prop	1235	Inb midc maneuver dV=90 m/s; done by main propulsion system
	[711]	**Tot Mars dep stg prop load**	**19648**	total at time of TMI burn
Earth capt	[561]	Earth arr stg usable prop tot	0	***Earth arr dV=0.0m/s; propulsive burn capture into 500 km by 24 hr ellip orbit***
	[704]	Earth arr stg prop residuals	0	2% residuals/reserve left after boiloff,burn and cooldown
	[700]	Earth arr stg 'cooldown' prop	0	3% post burn prop for reactor cooldown; no thrust/Isp counted in this approximation
	[562]	Earth arr stg outbound boiloff	0	983 day boff period;added boff from this tank also accounted in M capt &M dep boff
	[563]	**Total Earth arr stg prop load**	**0**	Total at time of TMI burn
	[570]	**Total combined prop load in aft tank**	**54811**	Mars capture/M dep prop: put in 1 tank; (along veh centerline aids NTR radiation attenuation)
Common tank				***1 continuous reinforced Silicon Carbide/Al metal martrix tank:*** dia:10.m, L:11.5 m,
	[683]	Single M dep/E arr tank wt	4142	filament wound; dens= 2436 kg/m3; 37ksi Wk. stress; tank skin thickness = 4.0 mm
	[684]	MLI wt	1369	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x dens
	[685]	Vapor cooled shield wt	1081	2 VCS - 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[686]	Meteoriod shield wt	918	One 0.80 mm sheet of Al; comparsion: SSF plans 0.8 mm, Mariner 9 used 0.4 mm
	[687]	Propel line/valves wt	225	length =10 m,double wall stainless steel H2 prop line; density= 7833 kg/m3, t=0.8mm
	[689]	Mass growth wt	1934	25% wt growth for tank shell, MLI,VCS,meteor shield,prop line & attachment
	[565]	Sum of inerts:single tank	9669	Total for single tank with all tank related inerts.
	[566]	Total for 1 tank	9669	Overall tank fraction [571] = 15.0 %
	[572]	***Combined Mars capt & Mars dep & Earth arr tank set & prop load***	**64480**	Total for ' Mars capture/Mars dep/Earth arr tank set ' at time of TMI burn
	[171]	***IMLEO***	**605340**	

Mac Chart: M NTR 2005 Conj E arr wt Art g
Boeing vehicle synthesis model run #: marsntrmtv dat:178

/STCARM/bhd/31Mar90

Earth dep stg & Mars capt prop - 2005 Conj Art-g (Mars-g) NTR Veh

14 % tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 *6/5/90*

Earth dep stg wt	[586]	Earth dep usable propel tot	227150	***Earth dep dV: 4267m/s (includes 300 m/s g-loss); Isp = 925 sec***
	[701]	Earth dep prop residuals	4543	2% residuals/reserve left after boiloff, burn prop, and cooldown
	[697]	Earth dep burn 'cooldown' prop	6814	3% post burn prop for reactor cooldown: no thrust/Isp counted for this estimated %
	[705]	**Tot Earth dep stg prop load**	238507	
				2 continuous reinforced Silicon Carbide /Al metal matrix tanks: dia: 10.0 m,
	[668]	Single tank wt (cyl/ellip ends)	7834	L:24.1 m, dens= 2436 kg/m3; 37ksi wk. stress, thickness = 4.0 mm, root 2 ellip ends
	[669]	MLI wt	2589	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x
	[670]	Vapor Cooled Shield wt	2045	density2 VCS: at 2 x 0.13mm Al outer sheets with 0.57 kg/m2 honeycombcore
	[671]	Meteoriod shield wt	1736	eachOne 0.80 mm sheet of Al; comparison: SSF uses 0.8 mm, Mariner 9 used 0.40
	[i312]	Tank/frame attachment	400	mmTank attachment mounting brackets & hardware as well as tank release
	[672]	Tank feed prop line wt	159	mechanism Short prop line from tank to main prop line:double wall, stainless steel:
	[674]	Mass growth wt	3691	10 meter 25% wt growth for tank inert,MLI,VCS,meteor shield,proplines, tank/veh
	[588]	Sum of single tank inerts	18455	attachment Total single tank inert wt;
	[592]	Total for 2 tanks	36911	Total for ' Earth dep tank set ': inert wt; Overall tank fraction [593] = 13.7 %
	[597]	***Earth Dep stage tot wt***	275418	Total Earth dep stg weight at time of Trans Mars Injection burn
	[610]	Mars arr usable prop tot	28816	***Mars arr dV: 863 m/s; eng Isp=925,*** H2 density = 70.8
	[702]	Mars arr prop residuals	576	2% residuals /reserveleft after boiloff, burn prop, and cooldown
	[698]	Mars arr burn 'cooldown' prop	864	3% post burn prop used for reactor cooldown; prelim;based on Westingh. estimate
	[611]	Mars arr stg outbound boiloff	1673	Boiloff for given MLI,VCS and Outb trip time; based on Boeing's 'CRYSTORE' prog
	[121]	Outbound midcourse prop	3235	Outb midc maneuver dV = 120 m/s; done by main propulsion from M arr tanks
	[612]	**Tot Mars arr stg prop load**	35164	
Mars arr stg tank *not used*	[633]	Single M arr tank wt	0	***0 SiC/Al metal matrix tanks;*** dia:10.0 m, L:0.0 m, dens=2436 kg/m3;thick=4.0mm
	[634]	MLI wt	0	MLI: density = 32 (kg/m3); 200 layers (4 inches) at 20 layers/cm
	[635]	Vapor cooled shield wt	0	2 VCS: at 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[636]	Meteoriod shield wt	0	One 0.80 mm sheet of LiAl: assumption-LEO assembley in protective hanger
	[i312]	Tank/frame attachment	0	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[637]	Tank feed prop line wt	0	Double wall, stainless steel 10 meter H2 propellant line;dens= 7833 kg/m3, t=0.8mm
	[639]	Mass growth wt	0	25% wt growth for tank inert,MLI,VCS,meteor shield,prop line&tank/veh attachment
	[614]	Sum of single tank inerts	0	Total for single tank, with all inerts.
	[615]	Total for 2 tanks	0	Overall tank fraction [620] = 14.5 %
	[617]	***Mars Arr stage wt [not used] Mars capt prop in single aft tank w M dep & E arr prop***	0	

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Mac chart: M NTR 2005 Conj M arr Art-g
Veh synthesis model run #: marsntrmtv.dat;178

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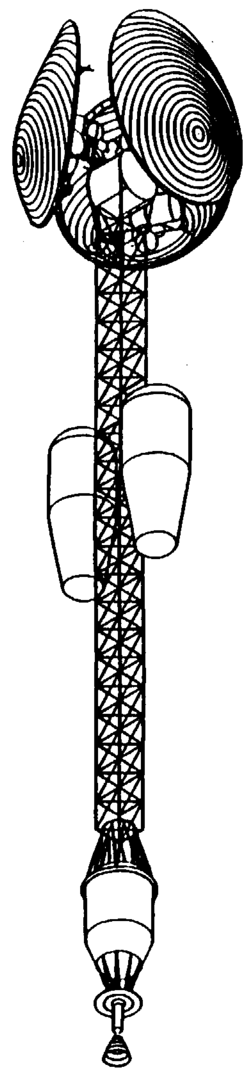
Exploration Emphasis Three Lander Art-g Conj Class NTR Veh

20 t or 1 t to Mars orbit, 1/3 g, crew of 6 (MEVs land 4), ECCV ret, Ref NERVA: Isp=925

dV's: TEI = 3900 m/s, MOC = 1530 m/s, TEI = 920 m/s, EOC Vhp = 5525 m/s, midcourse correction burns: outb=40 m/s, inb=40 m/s



Crew return via ECCV, no vehicle reuse

	Element / P/L taken & left in Mars orbit:	MEV type: Ref 20 t	Ref 1 t	Mini 1 t
[i313]	MEV descent only aerobrake	7000	7000	6000
[63]	MEV ascent stage	22464	22464	37366
	MEV descent stage	18659	18659	**n/a
[i66]	MEV surface cargo	25000	25000	5000
[i339]	***MEV total***	**73118**	**73118**	**48366**
[106]	***x 3***	**219350**	**219350**	**145098**
[378]	MTV crew hab module 'dry'	34790	34790	34790
[382]	MTV consumables & resupply	18270	18270	18270
[165]	MTV transit science	0	0	0
[381]	***MTV crew habitat module total***	**53060**	**53060**	**53060**
[i356]	Payload taken & left in Mars orbit	20000	1000	1000
[158]	MTV stg frame, struts, & misc inerts	4521	4521	4521
[792]	MTV stg RCS hardware & tankage	2034	1991	1834
[518]	NTR total engine wt	9684	9684	9684
[543]	NTR radiation shadow shield wt	4500	4500	4500
[118]	MTV nominal maneuver RCS prop	503	500	470
[761]	*MTV Art-g RCS spinup/down prop	7216	6851	5539
[121]	Outbound midcourse correction prop	1889	1784	1345
[122]	Inbound midcourse correction prop	431	428	402
[765]	Earth Orbit Capture (EOC) prop	n/a	n/a	n/a
[769]	Trans Earth Injection (TEI) prop	14280	14202	13404
[770]	Mars Orbit Capture (MOC) prop	72066	68104	51544
[657]	MOC/TEI common aft tank wt (1)	14564	14026	11711
[771]	***MTV propulsion/frame/propel tot***	**131690**	**126590**	**10450**
[705]	Trans Mars Inject (TMI) prop	274530	259430	196520
[592]	TMI side tanks wt (2)	41582	39622	31456
[597]	***TMI stage total***	**316110**	**299050**	**227980**
[230]	ECCV	8000	8000	8000
[171]	***IMLEO***	**748200**	**707040**	**535590**

,270,271,272

* Art-g prop for 5 spinup/spindown maneuvers
** single stage veh; aeroshell & landing legs left

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V. Support Systems

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Support Systems for the Mars Nuclear Thermal Rocket Vehicle.

The support systems necessary for the Mars Nuclear Thermal Rocket Vehicle are very similar in nature to those of the Mars Cryo/Aerobrake Transfer Vehicle. The discussion provided for the latter vehicle also applies generally for the NTR; however, detailed analysis for the specific systems needed to support the NTR have not been completed. It is currently assumed that this study will mainly consist of only deltas from the Cryo/Aerobrake Vehicle. Some manifesting work has been done for the major components of the NTR (as given on the following pages) using two different HLLV scenarios (each assumes the integrated aerobrake "Ninja Turtle" launch concept):

1) 10 meter x 30 meter shroud, 140 metric ton payload capacity

2) Mixed fleet consisting of:
   a) 7.6 meter x 30 meter shroud, 120 metric ton payload capacity; and,
   b) 10 meter x 30 meter shroud, 84 metric ton payload capacity

The total number of assembly missions for Scenario One is 7, while Scenario Two requires 9 flights. For the mixed fleet option, only the first and possibly last two assembly missions utilize the 120 mt payload carrier. This is due to NTR launch packages being limited as much by volume as by mass. Scenario One and Two also differ in that the first assumed that the MTV habitat should come up early (to assist in man-tended assembly operations) while the second delayed the MTV habitat until Mission Two (for use in ground test and verification).

Due to the mass of LH2 propellant required for the NTR trans-Mars injection, these tanks could only be partially full at ETO launch. The payload mass limitation of the Scenario One HLLV resulted in 26.6 mt of offloaded propellant which was carried to orbit on Mission Seven (this assembly flight may use a smaller ETO vehicle depending on tanker design). The Scenario Two (B) HLLV required a propellant offload of 152.1 mt. This offloaded propellant was carried up on Missions Eight and Nine and may be accommodated by either the (A) or (B) HLLV. These manifests assumed an Earth-based cryo tanker for "topping off" the NTR tanks; however, an on-orbit cryo depot is another option which is currently being studied.

The manifests given within have not yet been based on detailed ground processing and on-orbit assembly analyses. The philosophies and facilities chosen for ground operations (test and verification plans, payload processing, integrated assembly & checkout facilities, etc.) and assembly operations (Assembly Node location and capabilities, robotic and man-tended provisions, etc.) will obviously mature this manifesting.

Both the NTR and the Nuclear Electric Propulsion (NEP) vehicles have the added constraint of nuclear safe orbit considerations.. The nuclear safe orbit (NSO) has been customarily set at 800 km for 300 year life. The trade of whether to assemble the NTR at NSO or to build it at a lower orbit has not been completed; however, access to SSF, minimal assembly ΔV requirements, and natural radiation protection afforded by Low Earth Orbit assembly indicate this to be a favorable choice. For NTR, the amount of fission products produced even after a full Mars mission is about 250 grams; "cool" enough to do operations in as little as 20 km from the Space Station. This is closer than the debris environment constraints of 150 km from the Space Station.

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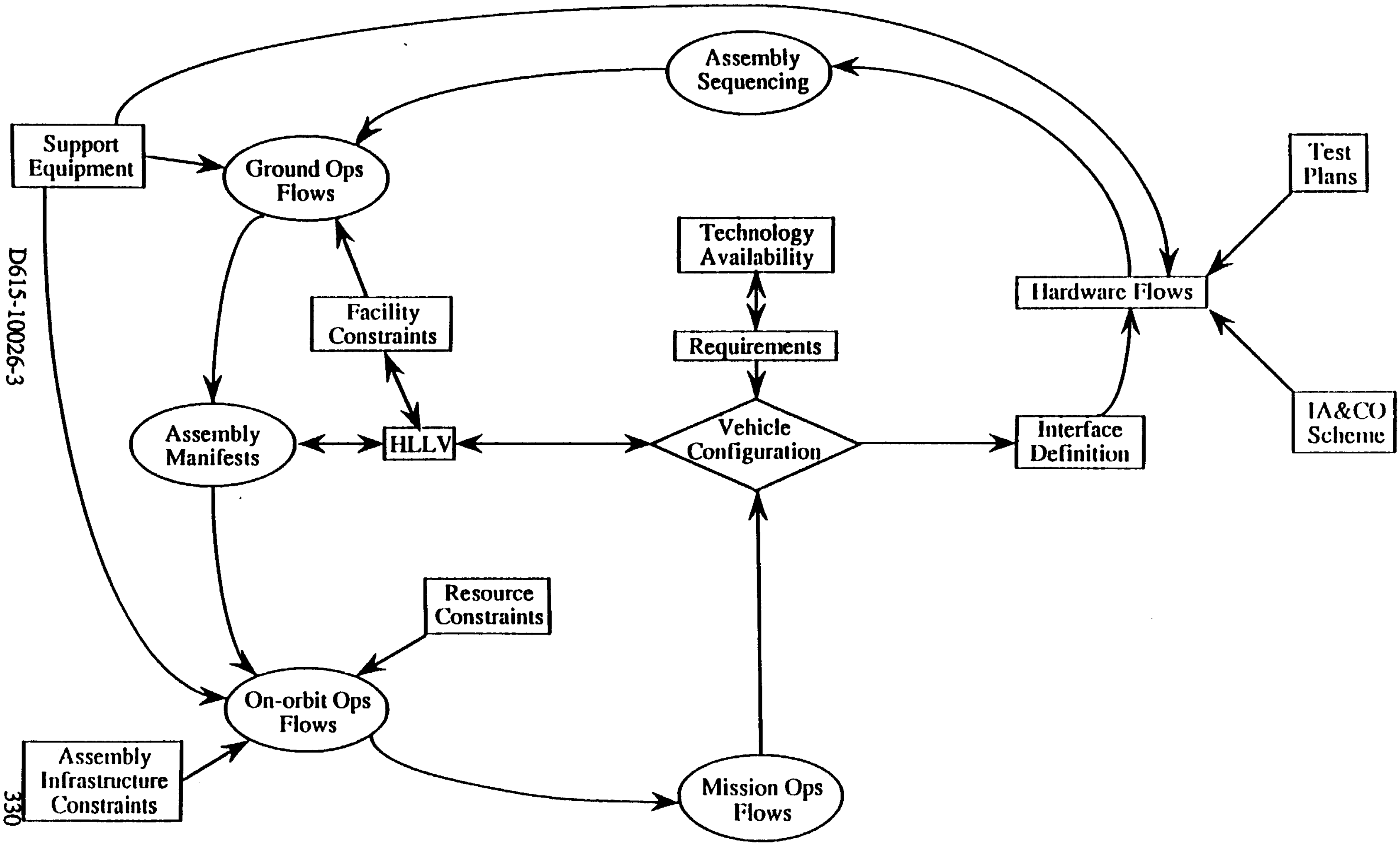
Operations Analyses and On-Orbit Assembly Concepts for NTR, NEP, and SEP

- Groundrules and Assumptions
- Assembly Node Concepts
- Manifesting/Packaging
- Assembly Flows
- Ground Processing
- Summary

Manifesting and Assembly Operations

This is an iterative, interdependent process

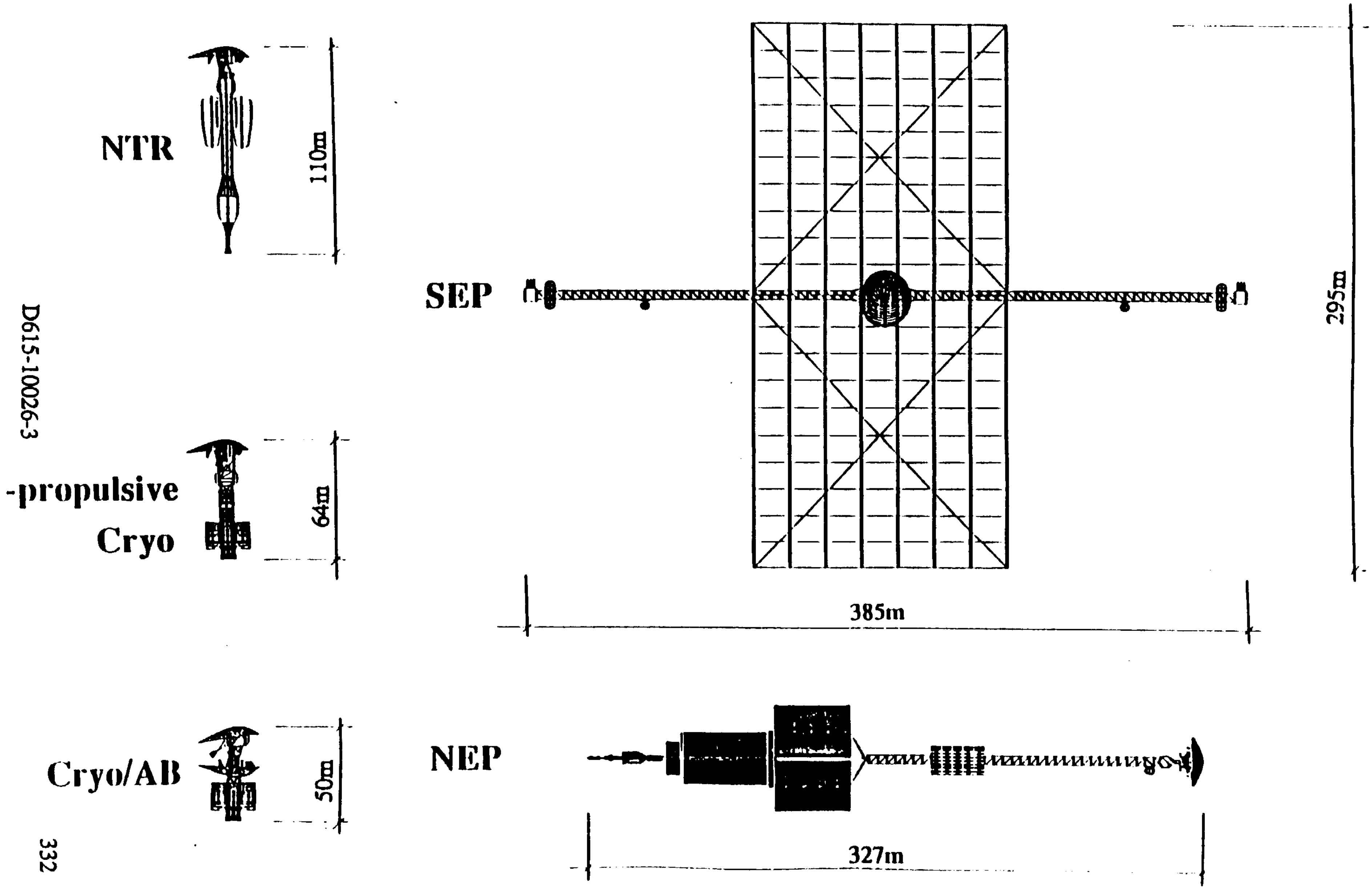


Manifesting and Assembly Operations- continued

Generic Assumptions and Ground Rules

- Based on Mars Vehicle (NEP, SEP, and NTR) configurations as of 3rd Quarterly Briefing with updates through 8/15/90
- Baseline Earth-to-Orbit (ETO) Vehicle (HLLV) has 10m x 30m shroud with 140 mt payload capability
- HLLV nosecone has some additional TBD volume for launch element packaging
- Nominal 85% payload packaging and mass factors used for HLLV manifesting (propellant tanks may be excepted)
- HLLV has a nominal 3 to 7 day station-keeping ability
- HLLV unloaded piece by piece by Cargo Transfer Vehicle (CTV)
- Crew transported to Assembly Location from SSF via ACRV
- CTV will be designed to support all identified manned/unmanned operations (on-orbit refueling may be available via on-orbit depot, HLLV provisioning, the Mars Vehicle itself, or SSF)
- HLLV launched on 90 day centers = time constraint for on-orbit assembly operations
- All Mars Vehicles are assumed to be launched February 2016
- Any localized debris shielding is removed from Mars Vehicle prior to departure from Earth (micrometeoroid shielding is assumed to be needed for the mission duration

Reference Vehicles
Size Comparison



On-Orbit Assembly Considerations

Required System/Service for Assembly	Available from Vehicle Itself ?
• Power	Yes
• Thermal Control	Yes
• Communications	Yes
• Micrometeoroid/Debris Protection	Possible with Localized/Temporary Shielding
• Reboost	Limited by Propellant
• Attitude Control	Limited by Propellant (gravity gradient should improve)
• GN&C	Yes
• Crew Volumes	
- Pressurized	Yes
- Unpressurized	Yes
• Robotics	Yes
• Test and Special Assembly Equipment	Yes
• Storage	Limited (but will be required for spares, etc.)
• Viewing/Proximity Operations	Undefined (but will be required)
• Consumable Resupply	Undefined (but will be required)
• SSF-compatible Interfaces	Limited (but will be required for crew transfer, etc.)
• Redundancy	Assembly-related Redundancy Undefined
• Disassembly/Refurbishment Accommodations	Possible Design Goal

Systems/Services Indicated as Available from Vehicle Exist *Only* After They Have Been Assembled

On-Orbit Assembly Considerations - continued

Vehicle's First Element Launch is Major Configuration Driver for Assembly Concept:

- No vehicle systems yet in place
- Even deployable vehicle systems require power and data
- Vehicle-independent HLLV "unloader" needed (this continues to be a need if HLLV is not brought to Assembly Site
- Vehicle-independent "assembler" may be needed
- Autonomous or external control of both HLLV and vehicle needed during assembly
- Constraints exist for assembly operation durations as well as for HLLV on-orbit lifetime

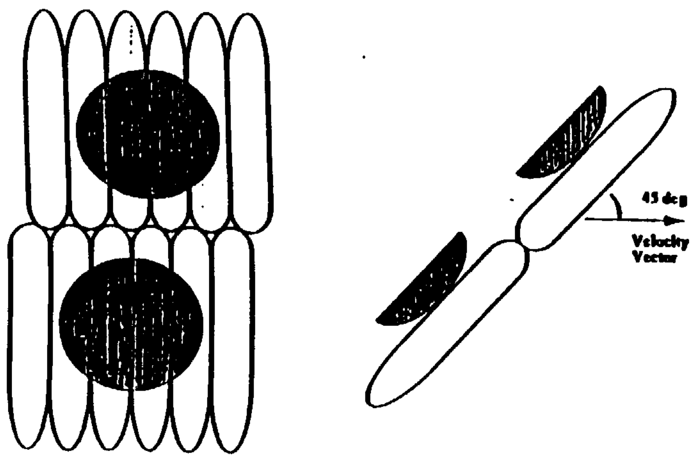
Assembly Mode Options:

- Robotic
- Ground-based Telerobotic + Automation
- Ground and Space-based Telerobotic + Automation
- **Ground and Space-based Telerobotic + Automation + EVA**
- Ground or Space-based Telerobotic + EVA
- Automation + EVA
- EVA

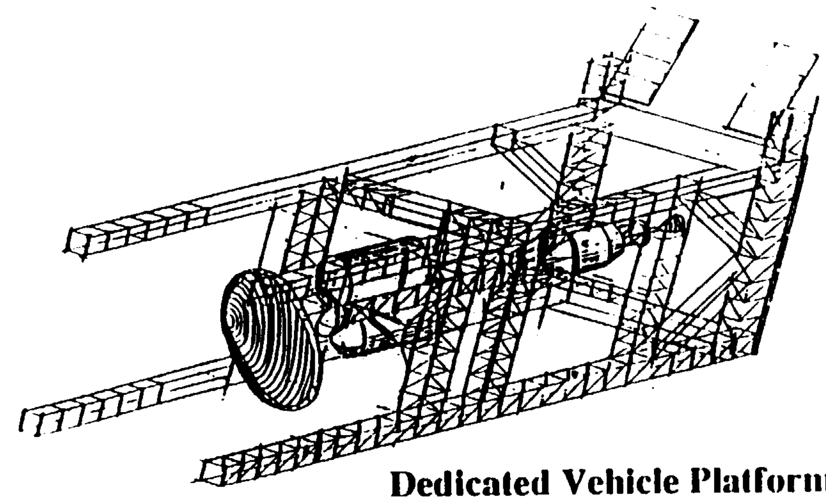
On-Orbit Assembly Concepts Summary

On-orbit Assembly Concept	Vehicle Applicability				
	CAB	CAP	NTR	NEP	SEP
• Vehicle as Its Own Platform	----	----	X	X	X
• ET-based Platform	X	X	MEV only	MEV only	MEV only
• Dedicated Vehicle Platform	X	X	X	?	----
• "I-Beam"	----	----	X	X	?
• "Smart" HLLV	FEL	FEL	FEL	FEL	FEL
• Flexible (Hinged) Truss	----	----	?	X	?
• Assembly Flyer	?	?	X	X	X
• SSF-based FEL Assembly	FEL, MEV (MTV)	FEL, MEV	FEL, MEV	FEL, MEV	FEL, MEV
• Tethered-Off-SSF Assembly	X	X	X	----	----

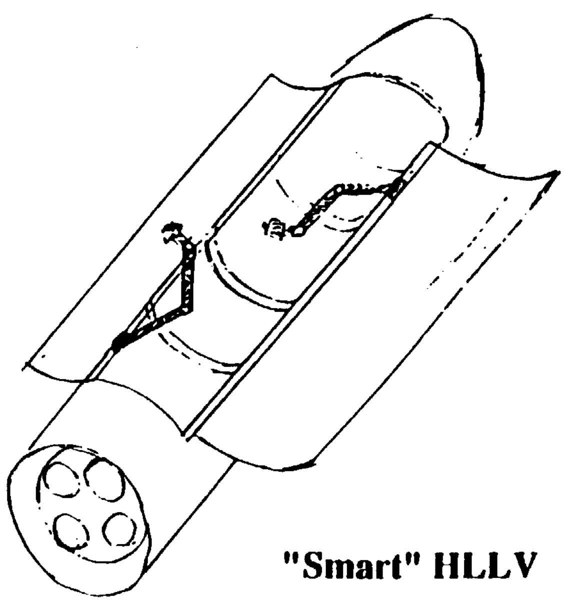
On-orbit Assembly Concepts



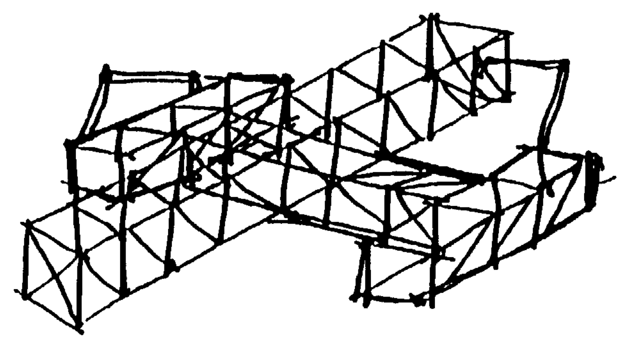
ET-based Platform



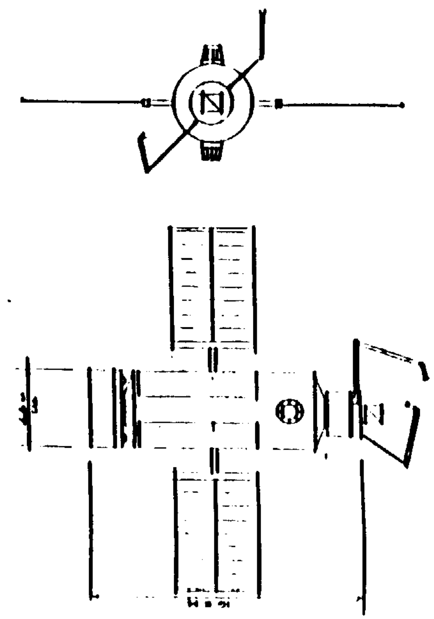
Dedicated Vehicle Platform



"Smart" HLLV

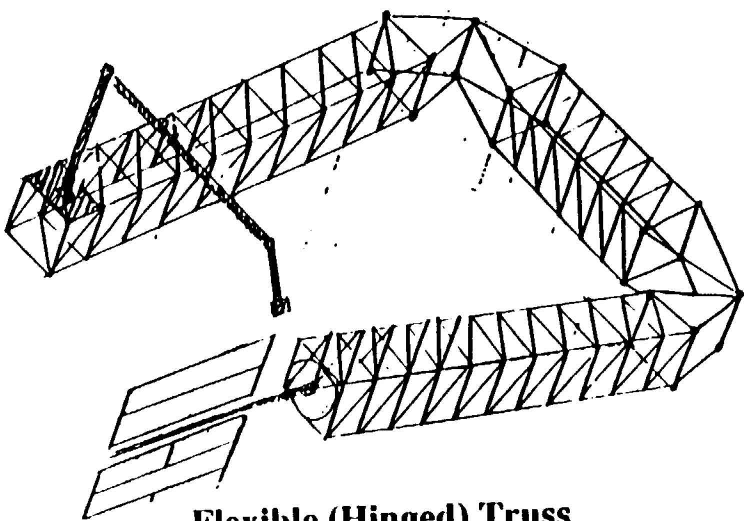


"I-Beam"

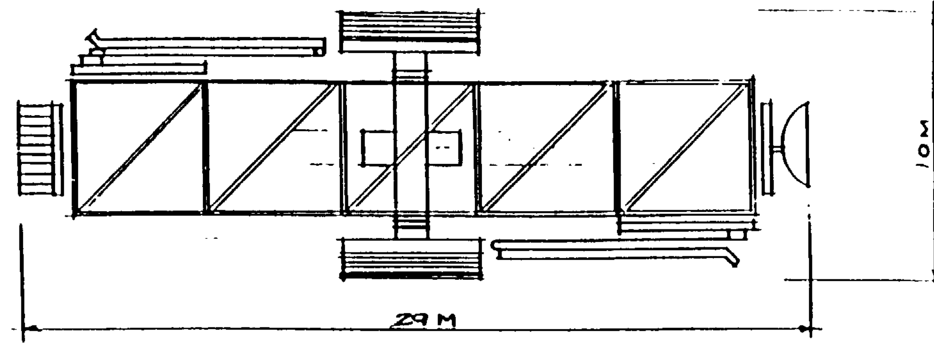


Assembly Flyer

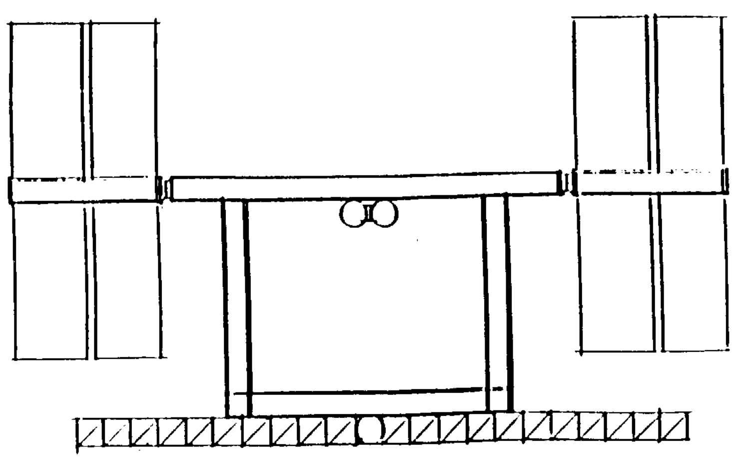
On-orbit Assembly Concepts - continued



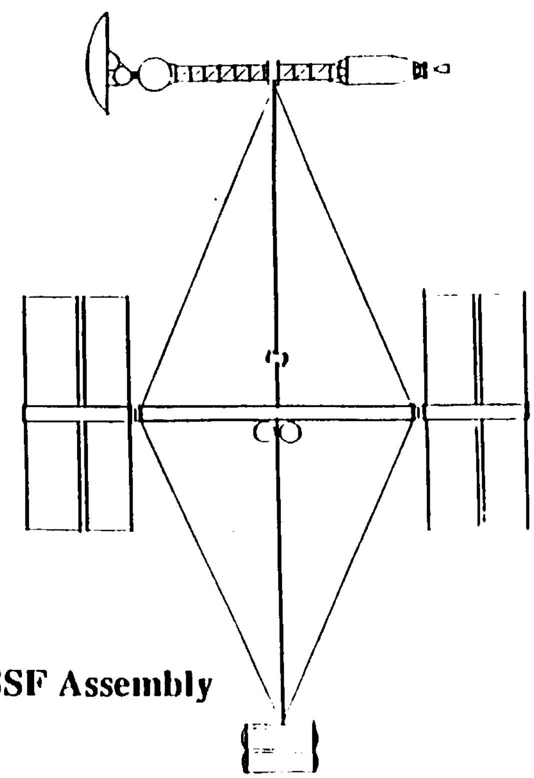
Flexible (Hinged) Truss



Vehicle as Its Own Platform



SSF-based FEL Assembly

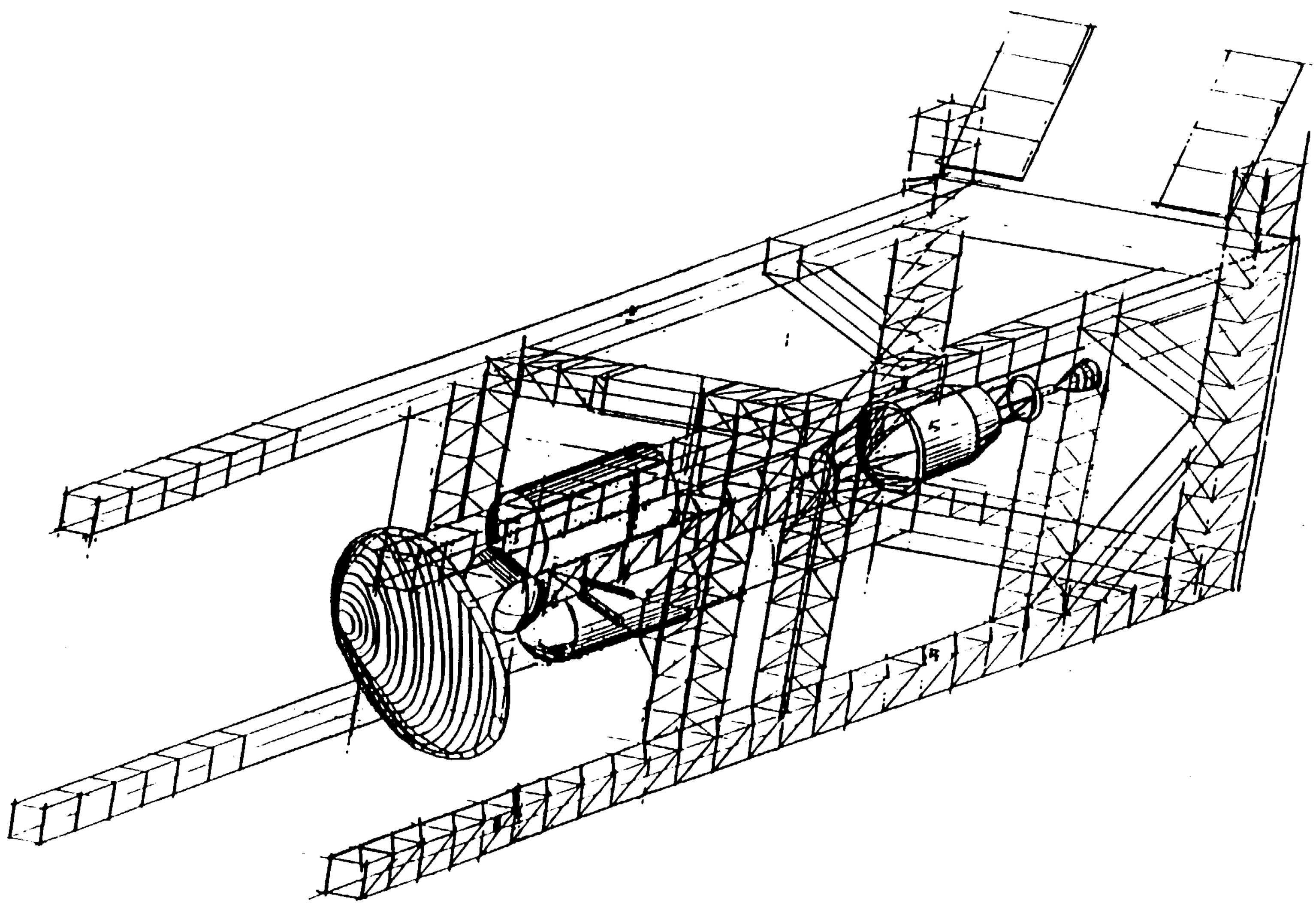


Tethered-Off-SSF Assembly

Dedicated Vehicle Assembly Platform Features

- Uses SSF type truss structure
- Dimensions 130m x 50m x 50m
- Movable and adjustable sections; can accommodate dual MEV configurations
- Aerobrake held from inside structure; TPS end is clear of obstructions. Allows unimpeded assembly and repair of TPS
- To release MEV from assembly platform, Aerobrake Assembly Section slides out longitudinally to the end of the platform, holding structure releases aerobrake, MEV moves out . MMV drops out from below the platform
- Pressurized Control Station with a logistics module and airlock
- Reboost system; occasional refueling needed and can be supported by CTV
- Gravity gradient stable
- Local debris shielding required
- Robot manipulator arms move longitudinally along tracks on platform truss
- Photo-voltaic arrays to provide power for platform and/or vehicle systems
- Storage fixtures are located along side the platform trusswork to store sections of the vehicle
- Platform can be controlled from SSF, from a ground station, and from the platform itself

Dedicated Vehicle Assembly Platform



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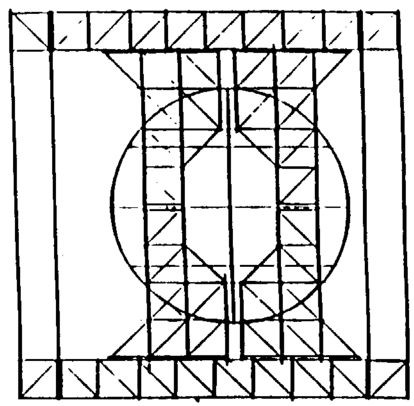
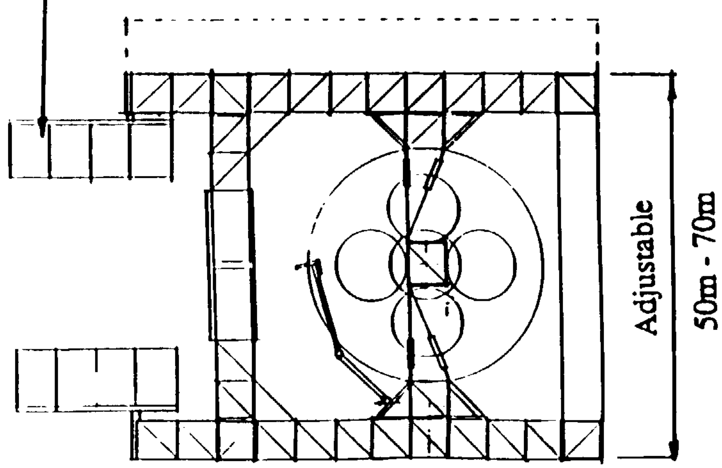
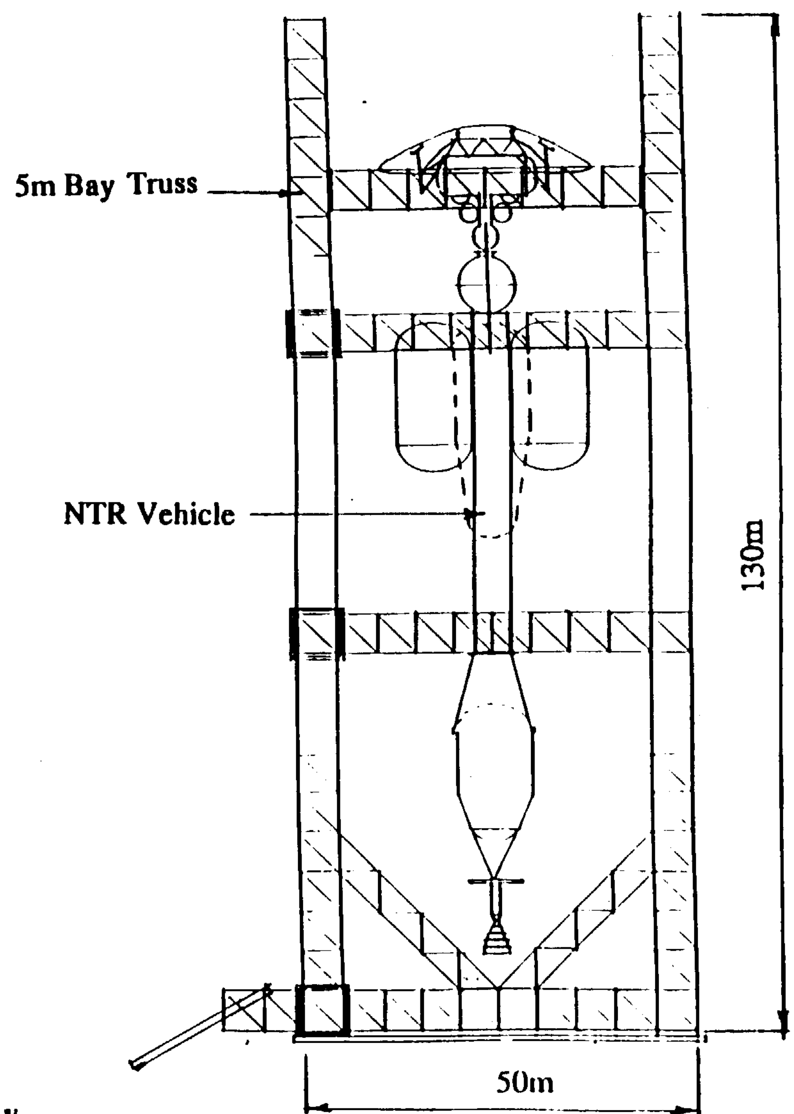
Dedicated Vehicle Assembly Platform



Aerobrake Assembly Bay



Tank/Truss Assembly Bay



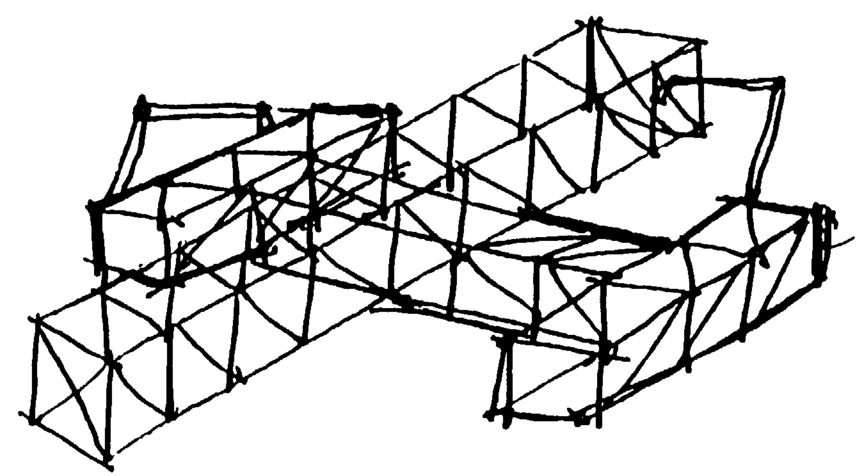
Side View

'STCAEM/crf/19Sept90

I-Beam Assembly Platform

- I-beam platform is carried up in first HLLV flight along with vehicle truss, both of which are self deploying
- I-beam platform attaches to one plane of vehicle truss
- Two robot arms that can move linearly on a base on side beams of i-beam platform
- Reboost, communication, avionics capabilities will be provided by vehicle being assembled
- Flies gravity gradient stable
- Debris shielding will have to be locally supplied to needed areas; minimum vehicle cross section facing debris
- "Pre"-assembly mission will be needed to set up vehicle and I-beam trusses (interfaces, cables, wires, conduits, communication, data, reboost, etc.) prior to main vehicle assembly

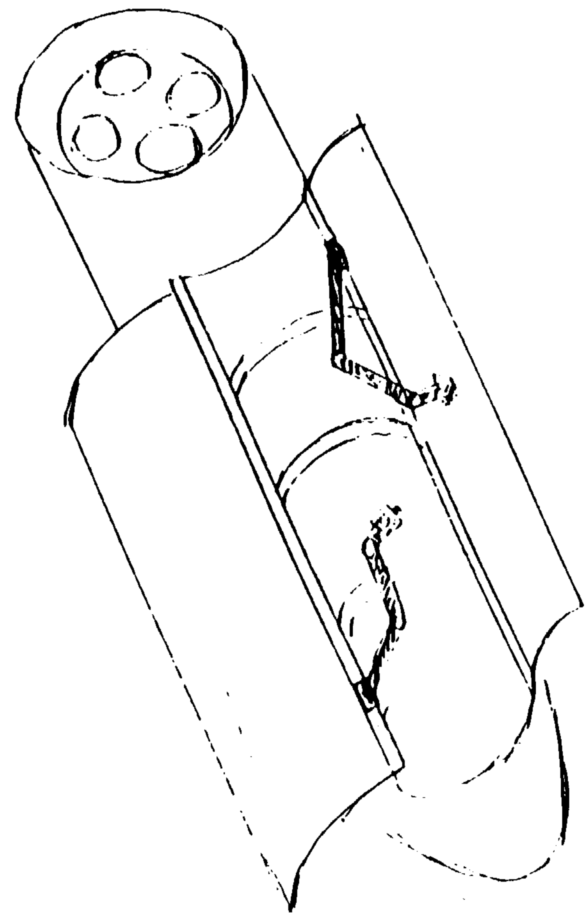
I-Beam Assembly Platform



"Smart" HLLV Assembly Platform

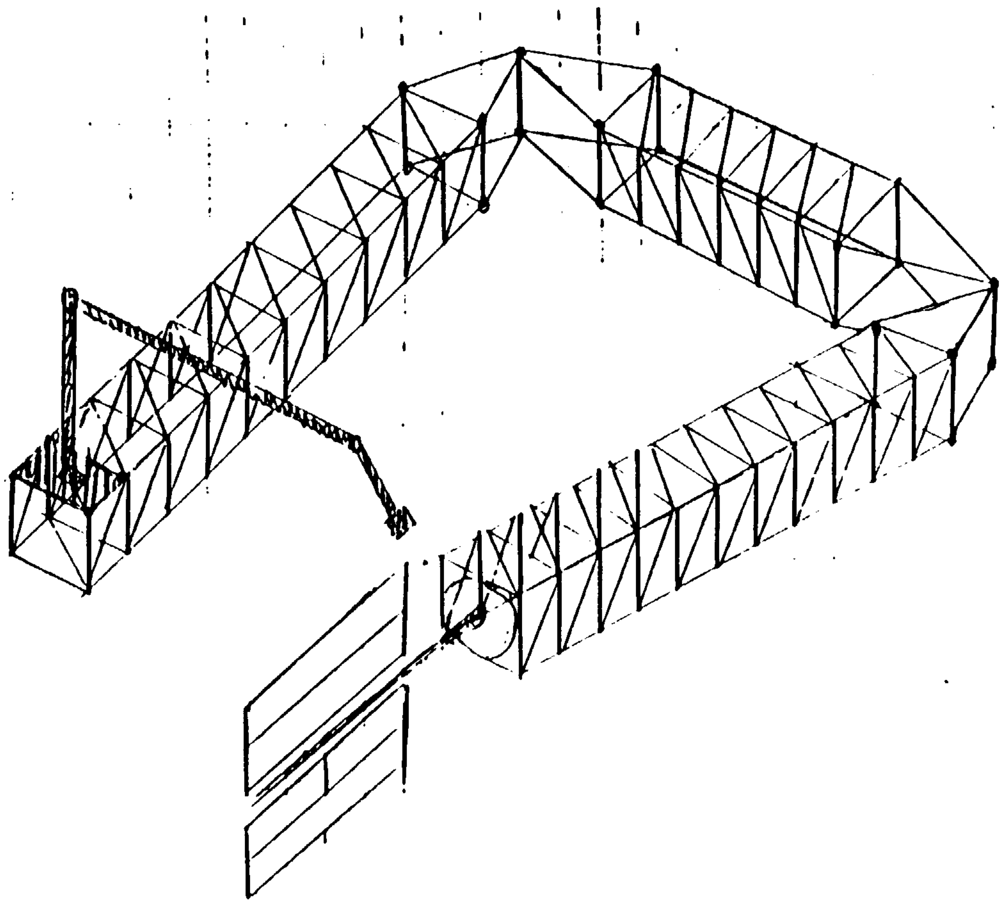
- Eliminates need for any additional platform
- Two robot arms similar to the current shuttle RMS, that can move lineraly on HLLV payload bay tracks
- HLLV provides partial debris shielding; supplemental local shielding will be required
- Telescopic mooring struts to attach vehicle to HLLV
- Reboost is provided by HLLV; refueling can be supported by CTV
- Vehicle's transit hab is used by crew during assembly operations
- All power for communication, avionics, robotics, RCS, etc., will be provided by vehicle systems
- "Pre"-assembly mission will be need to set truss interfaces, power, cables, wires, conduits, etc. Vehicle assembly proceeds after truss is readied for assembly operations
- Robot arms are transferred to vehicle from HLLV after a particular phase of assembly
- HLLV flies gravity gradient stable
- Only first assembly mission involves a "smart" HLLV; all others are cargo structures only

"Smart" HLLV Assembly Platform



Flexible (Hingeq) Vehicle Truss as Assembly Platform

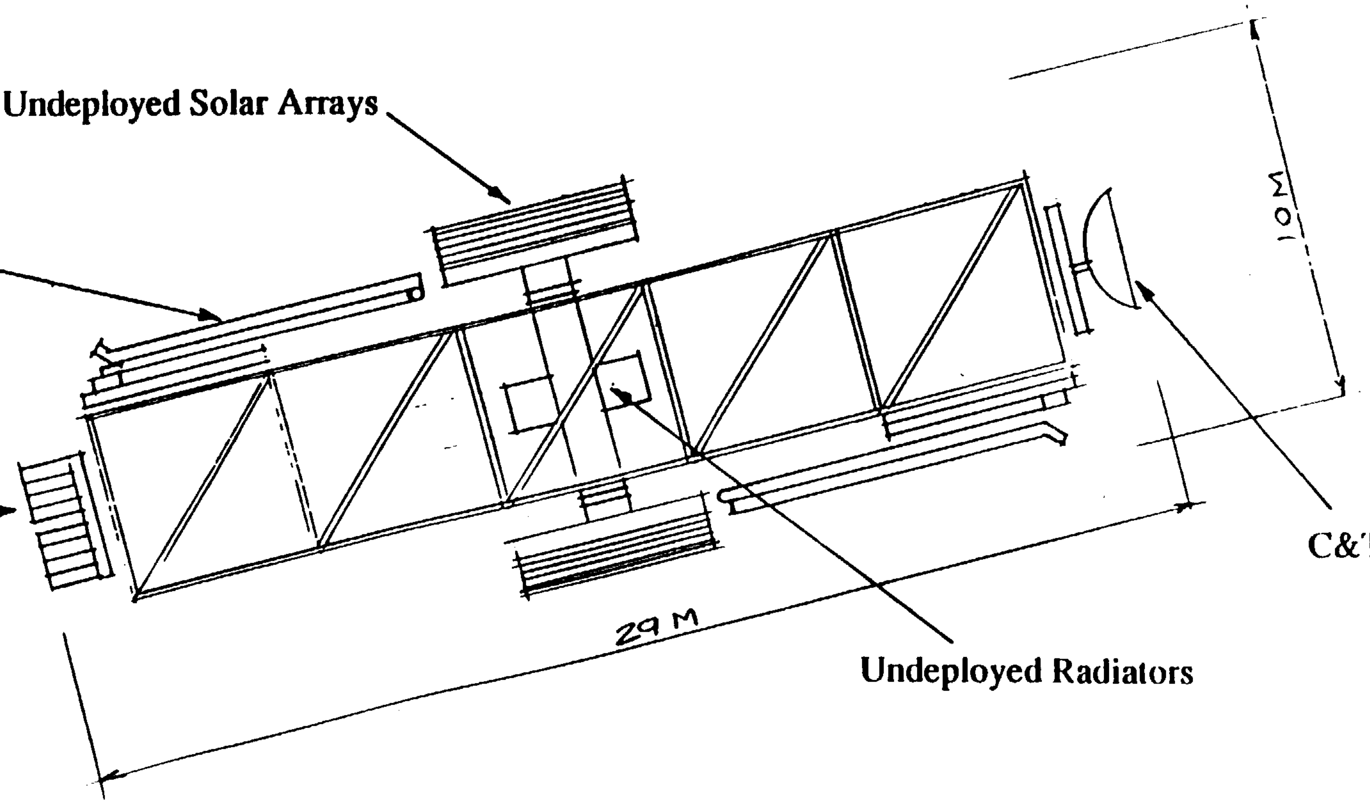
- NTR or NEP truss truss itself seves as assembly platform; truss can however flex at hinge points to provide reach behind the vehicle
- Minimum of two hinges to allow angular motion in one plane
- Eliminates need for any additional platform
- Two robot arms can be affixed to longest sections of hinged truss; robot arm can move along truss
- Hinges are modular and locking. Upon assembly completion, hinges lock and provide structural rigidity
- Local debris shielding required; vehicle is oriented such that minimum cross section faces debris
- Reboost is provided by vehicle's own reboost system with refuel support provided by CTV
- Vehicle's transit hab is used during assembly operations by crew
- All power for communication, avionics, robotics, RCS, etc., will be provided by vehicle's own systems
- "Pre"-assembly mission will be need to set up flex-truss, interfaces, power, cables, wires, conduits, hinge operation, communications data, reboost, etc.



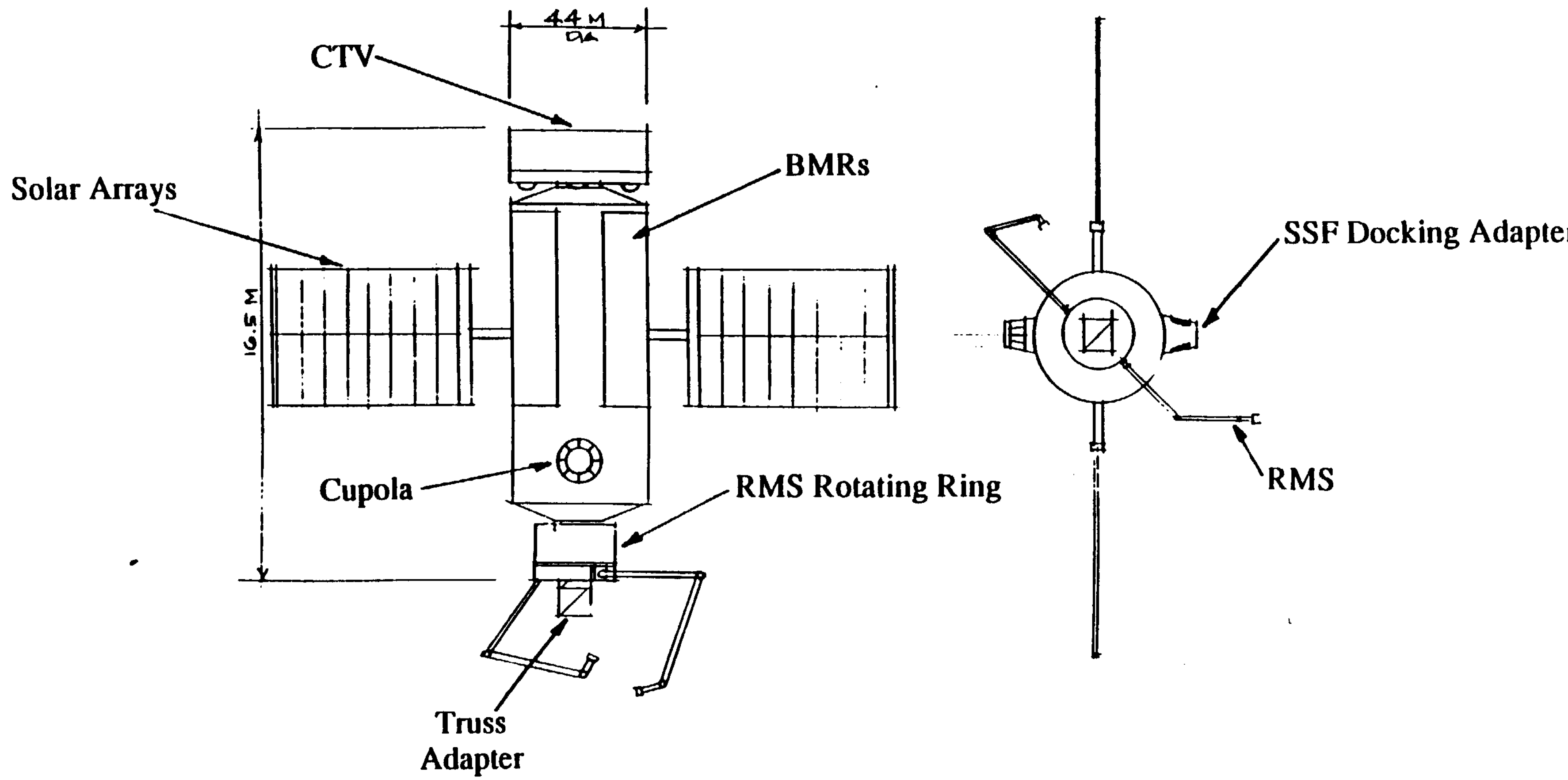
Using Vehicle as its Own Assembly Platform

- First Element Launch (FEL) delivers compact, fully integrated spacecraft:
  - Sized to be launched within 10m x 30m shroud
  - Contains self-sustaining and assembly support equipment necessary until the next element launch
  - FEL is integral part of the vehicle itself
  - HLLV releases automatically at proper attitude and orbit
  - On-board batteries deploy necessary power, radiator, and communication systems
  - Includes appropriate control and reboost systems

  These functions may also be accomplished by a CTV
  - Succeeding assembly missions are initially based from this element and expand with the vehicle
  - MMD shielding would be localized, integrated at launch, and removed prior to Earth departure



Assembly Flyer Concept



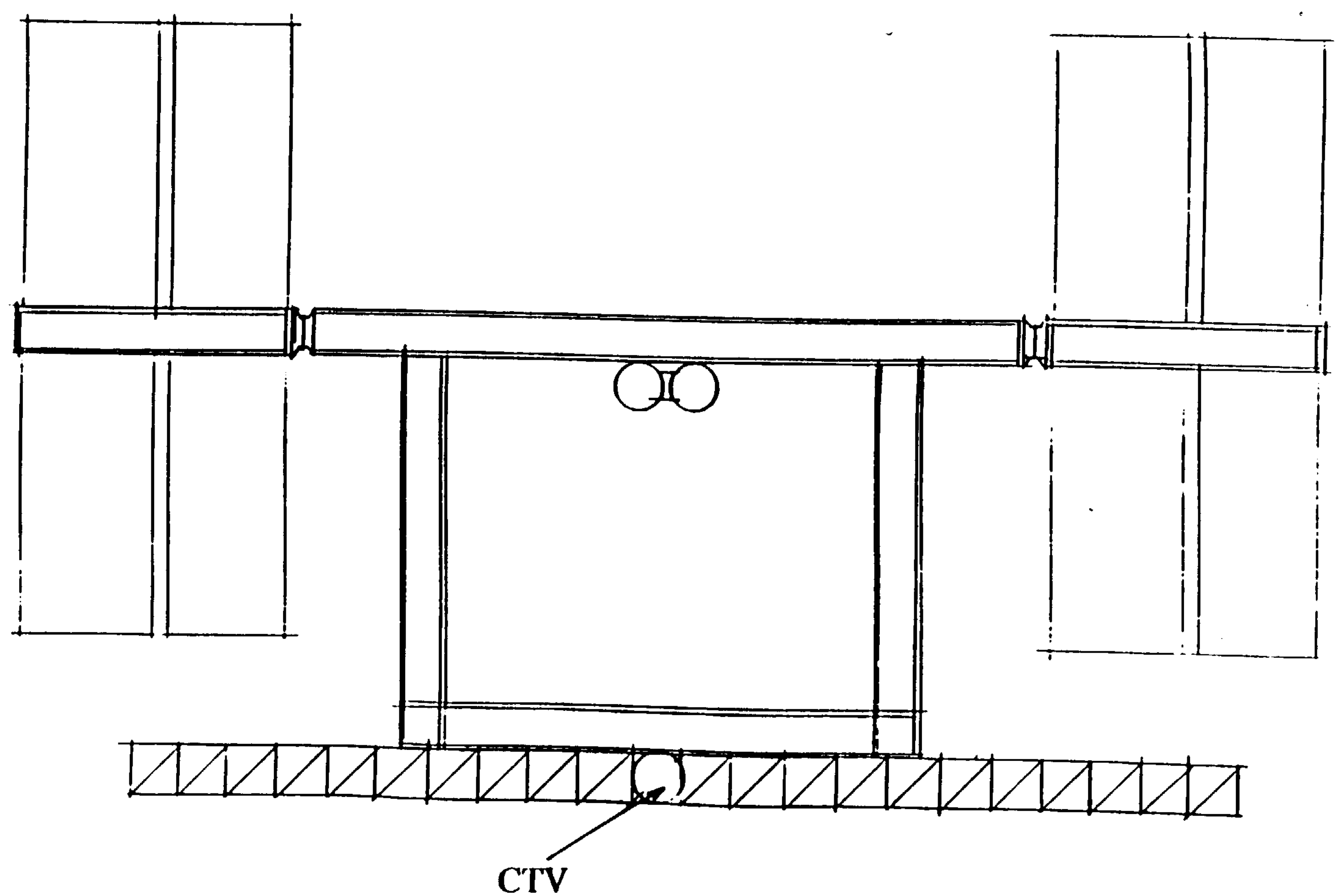
- Self-contained Assembly Flyer:
  - Performs assembly operations in any of three modes:
    - Free Flying: use for unloading HLLV, transfer of equipment/crew, etc.
    - Tandem Flying: use for handing off to vehicle, inspection, general assembly
    - Attached Operations: use for detailed and/or long duration assembly tasks (attachment may be directly to vehicle structure or to some temporary scaffolding)
  - Capable of manned and/or autonomous operations
  - Derivative from Industrial Space Facility (ISF)

SSF-Based Assembly of First Element Concept

- First Element of Mars Vehicle is assembled at SSF
  - Primary Truss
  - Power Systems
  - Thermal Control System
  - Communications
  - Avionics
  - Reboost and Attitude Control Systems
  - Remote Manipulator System
  - Utilities

- Once First Element is complete, the vehicle itself or a CTV docked to the vehicle transports it to an off-SSF location where remainder of vehicle is assembled:
  - Vehicle is enabled to assemble remainder itself
  - If needed, CTV aids with reboost and control until supplemental systems arrive
  - Debris shielding may be localized
  - MEV is assembled prior to Aerobrake/Aeroshell assembly
  - Temporary scaffolding may be used as needed

- First Element may be assembled with its orientation parallel (as shown) or perpendicular to SSF, depending on:
  - Drag effects
  - Controllability
  - Microgravity effects on SSF
  - RMS reach



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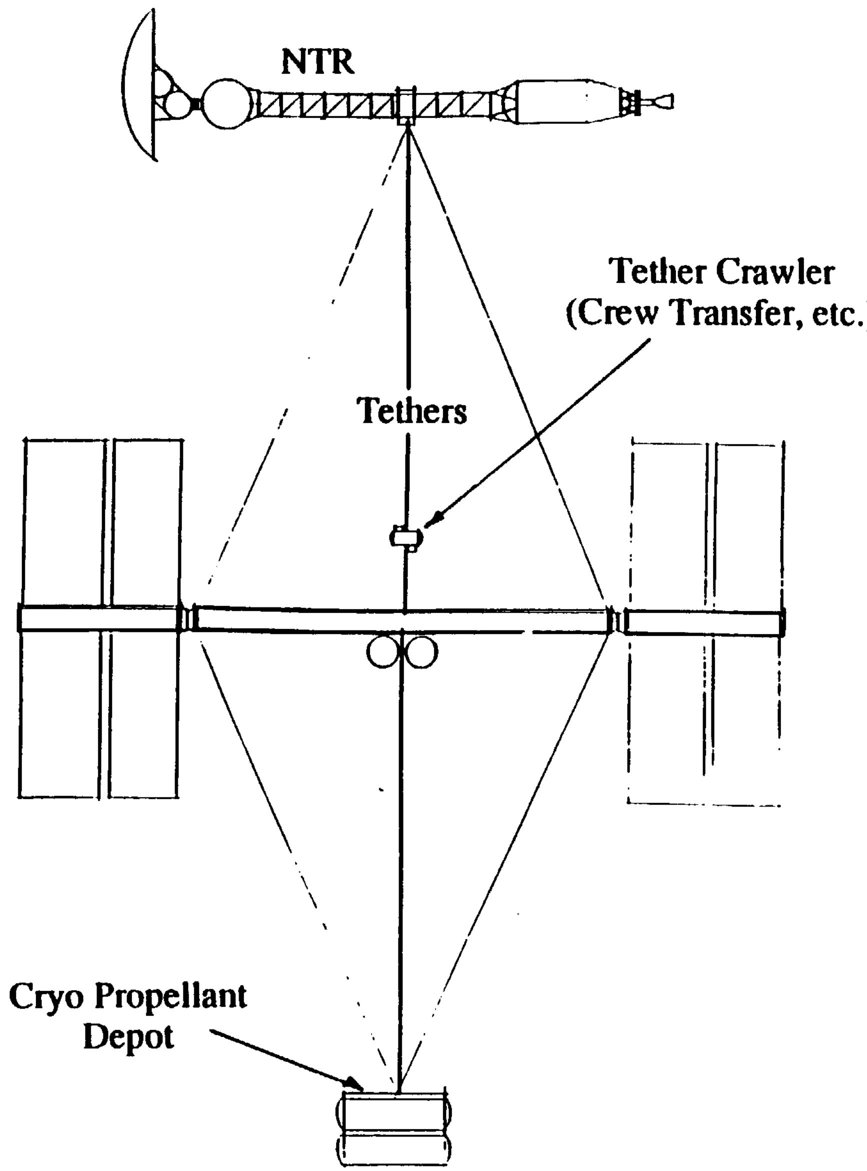
Tethered-Off-SSF Assembly Concept

- Allows easy crew and logistics access (pressurized or unpressurized) between SSF and assembly area
- Removes hazardous operations and materials from SSF
- SSF facilities (with upgrades) may be available to both vehicle and on-orbit depot:
  - Power
  - Data
  - Communications

  } via tether
  - Attitude and Reboost Systems
- Center of mass may be maintained in the SSF Labs by moving the vehicle and depot along the tether as the vehicle is built up, propellant is transferred, etc.
- Tether also serves to mitigate dynamic disturbances to SSF caused by assembly or propellant operations



Assembly Node Concepts Pros and Cons

Node Concepts	Key Features/Advantages	Key Disadvantages
Dedicated Assembly Node	• Abundant storage • Totally self-contained • Vehicle systems unused • Multiple robot arms • Sections of vehicle may be assembled simultaneously	• Larger than SSF • Will take long time to construct • Excessive reboost requirements • Mechanically complex • Local debris shielding required • Must be in place prior to vehicle assembly
I-Beam Platform	• Can be carried up in first HLLVflight • Can easily reach most parts of vehicle with two robot arms • Uses vehicle for comm., data, RCS, power after initial deployment • Can serve as base for experiments	• Fuel cells, batteries required for initial deployment • Limited storage area • Precursor mission required for deployment
"Smart" HLLV Platform	• No additional platform required • HLLV shroud provides limited debris shielding • HLLV provides for communication, data, RCS, GNC, etc. • Robot arms transferable to NTR	• Increased HLLV complexity • Reboost fuel has to be replenished • Limited storage • Vehicle must be detached from HLLV prior to assembly complete • Local debris shielding required
Hinged Truss Platform	• Uses vehicle truss as assembly platform; no other platform needed • Reach to remote engine section of vehicle provided by flexing truss at hinges • Vehicle subsystems used; no additional systems necessary	• Requires a precursor mission to deploy truss • Batteries, fuel cells necessary for initial deployment • Reboost, comm., data, power, must be in place prior to assembly start • Limited storage • Local debris shielding required
Vehicle as its own Platform	• Reduces needed on-orbit infrastructure • Deletes additional facilities and resources needed for designing, builiding, launching, and maintaining separate assembly platform	• Requires dedicated HLLV flight for non-optimized packaged first element • Requires vehicle to have additional control, reboost • No additional storage • Requires batteries or fuel cells for initial deployment • Requires localized debris shielding

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Assembly Node Concepts Pros and Cons
(continued)

Node Concepts	Key Features/Advantages	Key Disadvantages
Assembly Flyer Platform	• Performs HLLV unloading, payload/crew transport, and assembly with one vehicle • Compatible with SSF • Capable of manned/robotic operations • Uses CTV for main P/A • Can serve as free flying platform between assemblies	• No additional storage • Requires vehicle to have additional control and reboost systems • Requires development and production of sophisticated man-rated space vehicle • Requires localized debris shielding
SSF Based Assembly of First Element	• Uses planned SSF growth concept • Provides quick and easy crew logistics access to initial assembly operations • Allows verification and checkout of critical systems prior to independent vehicle operations • Does not disrupt SSF operations beyond first assembly mission (remainder of assembly based from vehicle itself)	• Impact to SSF (resources, microgravity, drag, etc.) • Eventually requires vehicle to have additional control and reboost systems • Requires localized debris shielding • No additional storage beyond first element
Tethered off-SSF Assembly Platform	• Compatible with current SSF design • Provides quick and easy crew and logistics access to entire assembly and propellant transfer operations • Microgravity and dynamic loads impacts to SSF minimized by tether • Removes hazardous operations and materials to SSF standoff distance	• Impact to SSF resources • Requires localized debris shielding • No additional storage • Requires additional reboost and control systems on SSF

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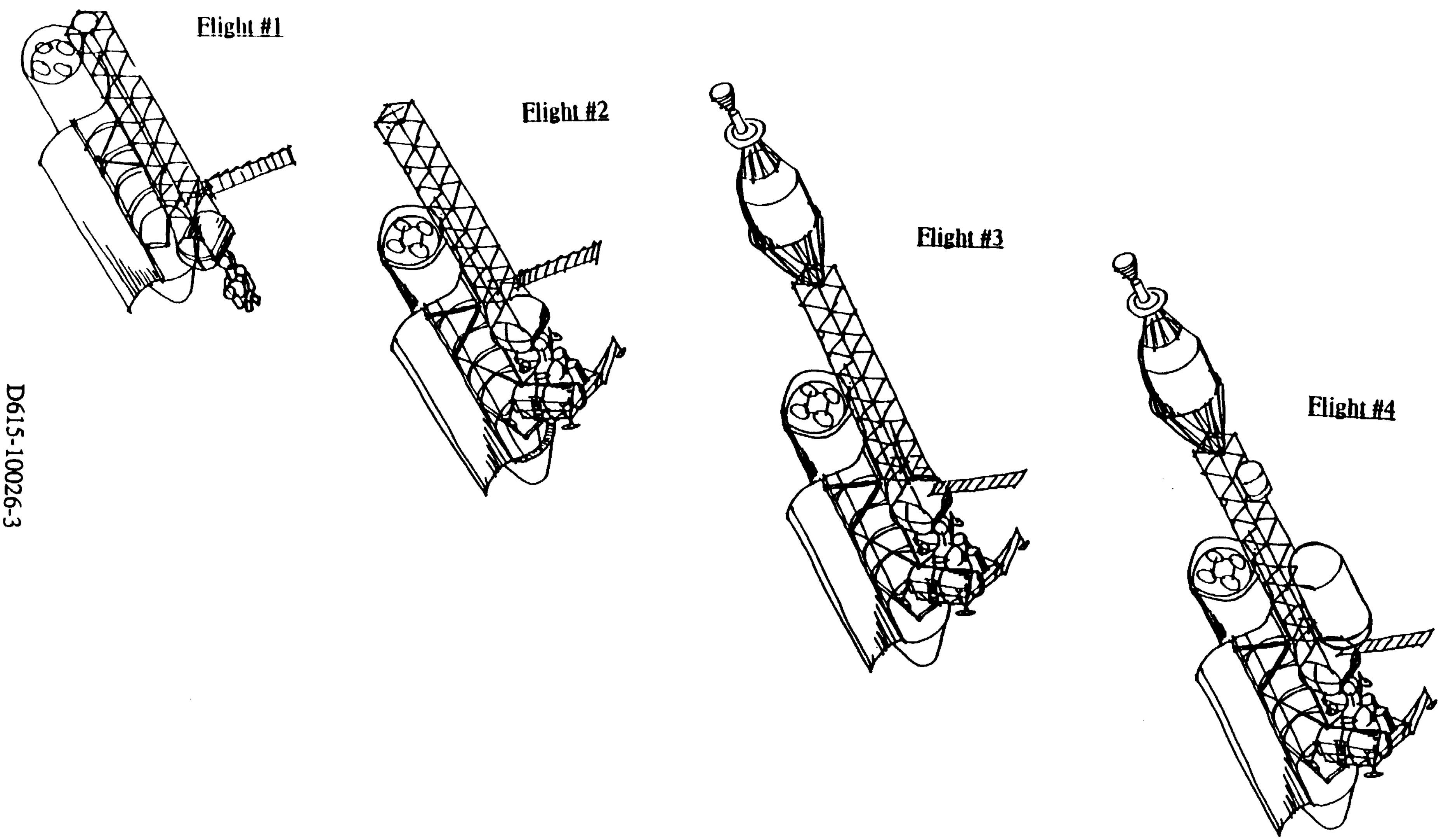
"Smart" HLLV Platform: NTR Assembly Assumptions

- Surface science equipment and rovers are packaged into MEV descent stage mission (2).
- Truss-to-EOC tank structure, tank-to-shield structure, and shield/engine are integrated to EOC tank. Engine nozzle is mounted in reverse and fits over the engine/reactor in order conserve space. Part of the throat of the nozzle protrudes into nose cone.
- "Off-loaded" fuel tankers are brought up with MOC tanks.
- After the fourth mission, the NTR vehicle is detached from the HLLV, the robot arms are transferred to the vehicle, and the HLLV is deorbitted. This detachment is necessary to enable assembly of remaining fuel tanks and aeroshell.
- If the aeroshell is brought up in sections, then two HLLV flights are dedicated to aeroshell assembly. Aeroshell can be assembled from MEV landing leg attachments. The sections of the aeroshell that include the robotic arms are brought up in the first flight as these will be utilized to assemble the aeroshell.
-

"Smart" HLLV Platform: NTR Assembly



ADVANCED CIVIL SPACE SYSTEMS

"Smart" HLLV Platform: NTR Assembly

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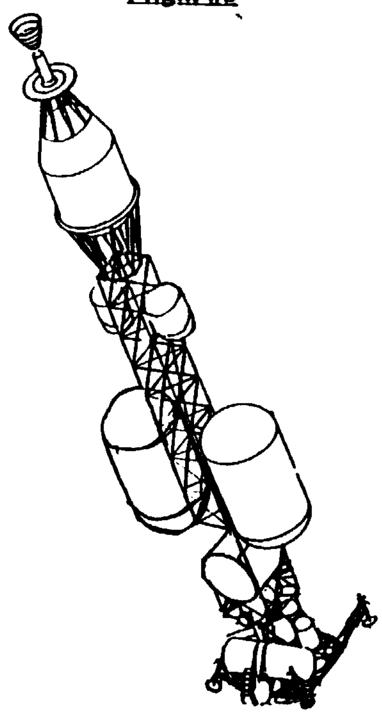
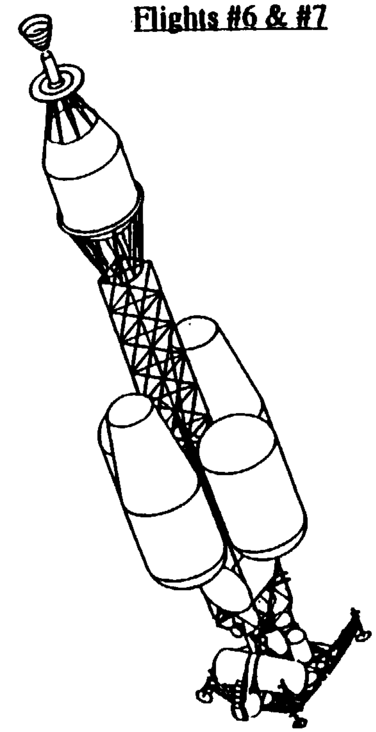
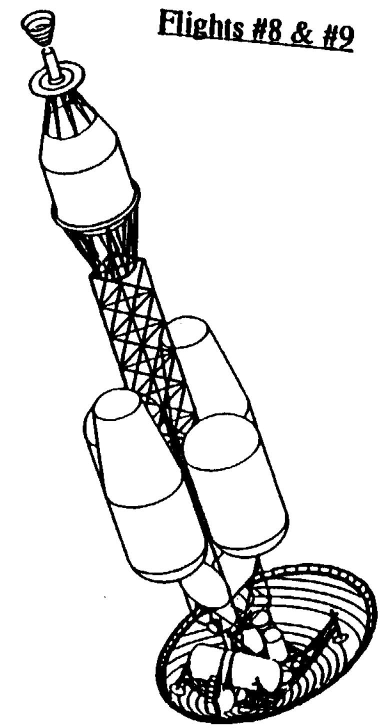
Flight #5



Flights #6 & #7



Flights #8 & #9

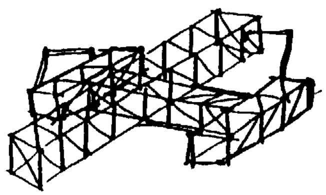


I-Beam Platform: NTR Assembly Assumptions

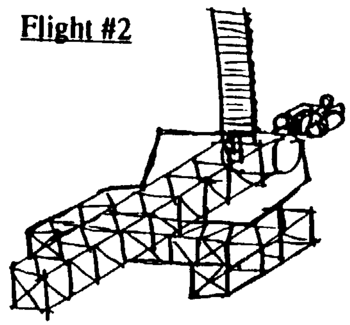
- Deployment of I-beam platform and the NTR vehicle truss (first HLLV mission) subsumes power, data, reboost, and communication facilities are available on the platform.
- This node concept utilizes NTR vehicle systems during assembly; these systems may have to be refurbished prior to departure.
- Communications, GNC, reboost, ACS, etc., that are part of the platform, are common to the NTR vehicle.
- Rovers and science packages are packaged into descent stage of MEV.
- MEV ascent and descent stages are not packaged into containers but are configured for stowage into HLLV. Descent stage landing legs may have to be disassembled.
- Aerobrake is brought up in sections and requires two flights of HLLV.
- Following assembly of the NTR vehicle, platform is detached from vehicle and could be used as a platform for experiments.

I-Beam Platform: NTR Assembly

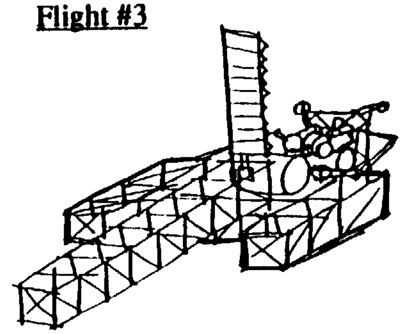
Flight #1



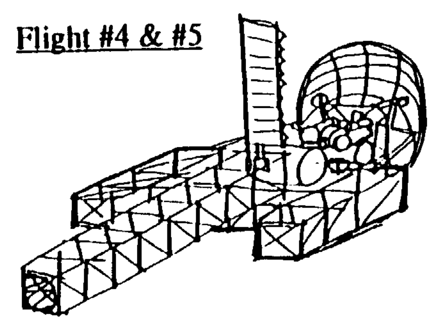
Flight #2



Flight #3

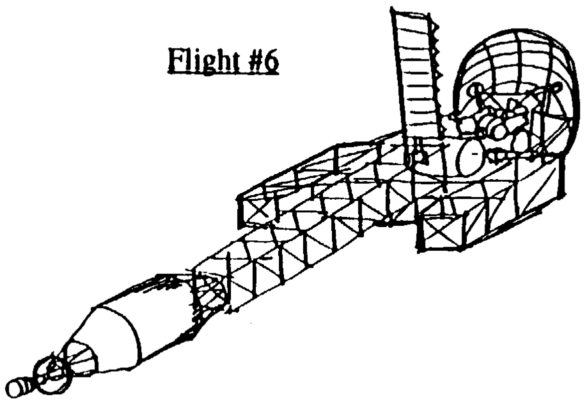


Flight #4 & #5

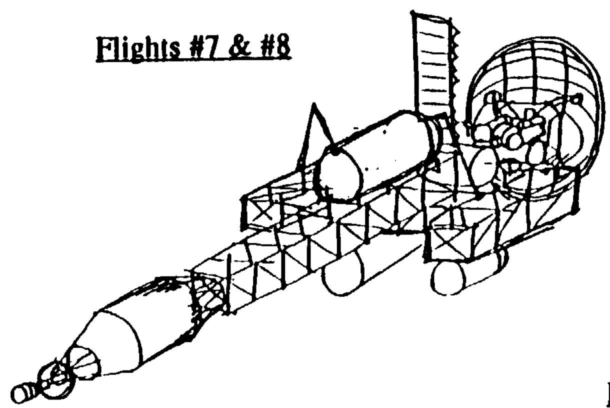


I-Beam Platform: NTR Assembly

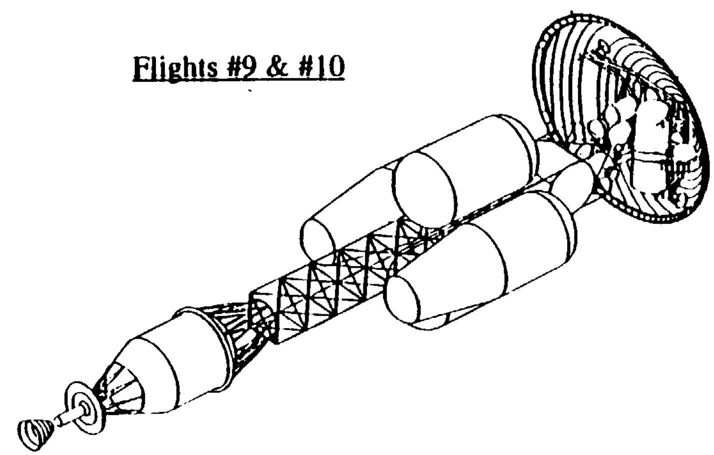
Flight #6



Flights #7 & #8



Flights #9 & #10



ADVANCED CIVIL SPACE SYSTEMS

NTR Component Manifest Data

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NTR Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• MEV			
Aeroshell	1	28 x 30 x 7 box	9.51
Descent System (incl 2 rovers)	1	9.5 x 20 x 4 box *	32.83
Ascent System	1	9.5 x 9.5 x 5.5 box	24.83
Surface Payload Module	1	13 x 4.4 (dia) cylinder	25.00
Surface Payload Module Airlock	1	2.9 x 3 (dia) cylinder	4.50
		Subtotal =	96.67
• MTV			
MTV Hab Module	1	10 x 8 (dia) cylinder	40.30
MCRV	1	3 x 4 (dia) cylinder	7.00
MTV-to-MEV Tunnel and Airlock	1	6 x 3 (dia) cylinder	7.00
Main Truss (includes RCS, RCS fuel, main fuel lines, mass growth, GNC)	1	7 x 7 x 7 box (deployable) *	5.6
Mars Orbit Capture Tanks (include fuel)	2	20 x 10 (dia) cylinders	178
Trans-Mars Injection Tanks (include fuel)	2	30 x 10 (dia) cylinders	329
In-Line Tank (includes fuel)	1	19 x 10 (dia) cylinder	101
Tank-to-Truss Structure	1	11.5 x 3	1.00

* These represent launch package dimensions, not mission configuration

NTR Component Manifest Data- continued

NTR Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• MTV (continued)			
Tank-to-Shield Structure	1	7 x 7	2.40
Shield/Engine	1	10 x 7 (dia)	11.80
		Subtotal =	683.10
		NTR Total =	779.77

NTR - Manifesting and Packaging

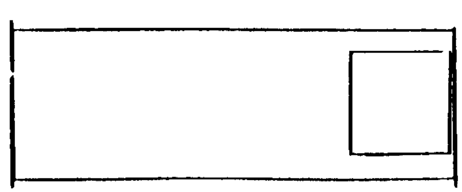
(10m x 30m Shroud, 140 mt HLLV using Dedicated Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) with 140 metric ton capability and 10m x 30m shroud
- Sequencing based on Dedicated Assembly Platform concept
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NTR configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assembled on orbit (in ten pieces) and requires two dedicated HLLV flights
- 50 mt of propellant, off-loaded from TMI tanks to two tankers, are carried up along with MOC tanks in missions five and six

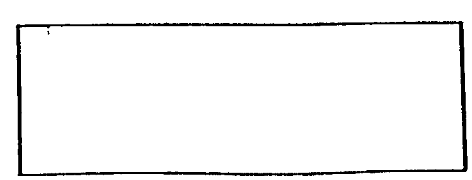
Assembly Mission One

- MEV Aeroshell (4 out of 10 pieces)
- NTR Truss



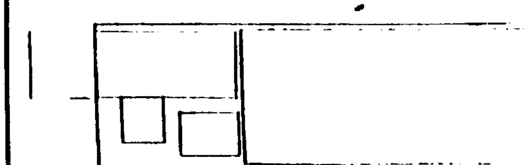
Assembly Mission Two

- MEV Aeroshell (6 out of 10 pieces)



Assembly Mission Three

- MEV Descent Stage (incl 2 rovers)
- Surface Payload Module
- Airlock for Surface Payload Module (integrated with surface module)
- MCRV

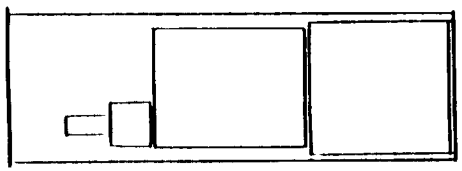


NTR - Manifesting and Packaging (continued)

(10m x 30m Shroud, 140 mt HLLV using Dedicated Platform)

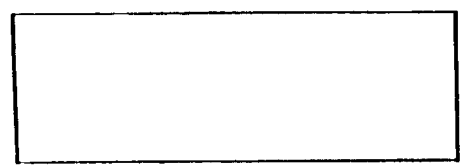
Assembly Mission Four

- MTV Hab Module
- MTV-to-MEV Airlock and Tunnel
- MEV Ascent Stage
- Equip ment for Communications, GNC, RCS (Reboost and Attitude Control)



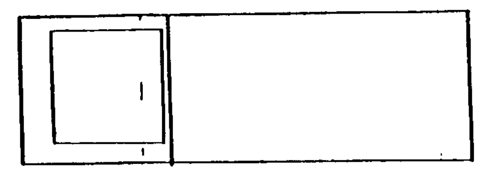
Assembly Mission Five

- TMI Tank + Fuel (1 of 2)



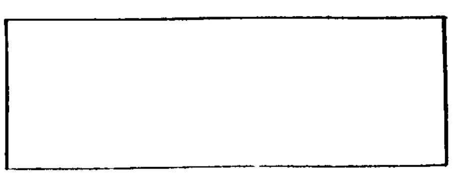
Assembly Mission Six

- Mars Orbit Capture Tank and fuel (1 of 2)
- Off-loaded fuel tanker (1 of 2)



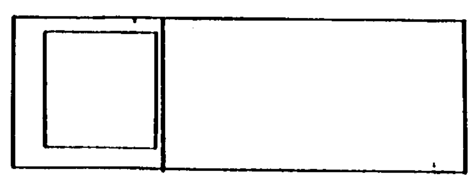
Assembly Mission Seven

- TMI Tank + Fuel (2 of 2)



Assembly Mission Eight

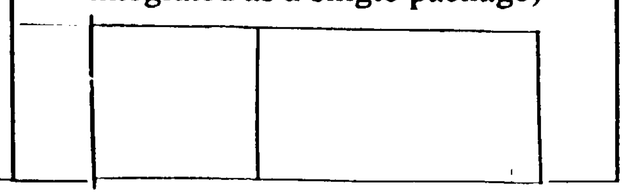
- Mars Orbit Capture Tank and fuel (2 of 2)
- Off-loaded fuel tanker (2 of 2)



Assembly Mission Nine

- In-Line Tank + Fuel
- Tank-to-Truss Structure
- Tank-to-Shield Structure
- Engine/Shield

(These components are integrated as a single package)



NTR - Manifesting and Packaging

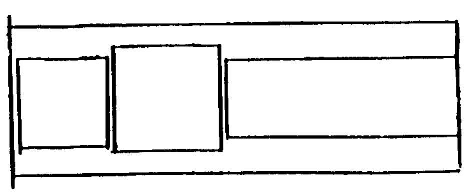
(10m x 30m Shroud, 140 mt HLLV using I-Beam Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) with 140 metric ton capability and 10m x 30m shroud
- Sequencing based on I-Beam Assembly Platform concept; platform is carried up in first assembly mission
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NTR configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assembled on orbit (in ten pieces) and requires two dedicated HLLV flights
- 50 mt of propellant, off-loaded from TMI tanks, to two tankers, are carried up along with MOC tanks in missions seven and eight

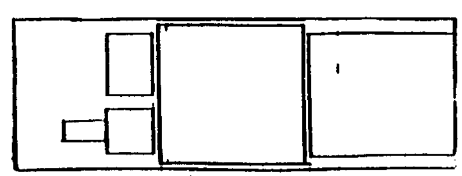
Assembly Mission One

- NTR Truss
- I-Beam Platform with 2 Robot Arms
- Communication, RCS (for reboost and Attitude control), GNC



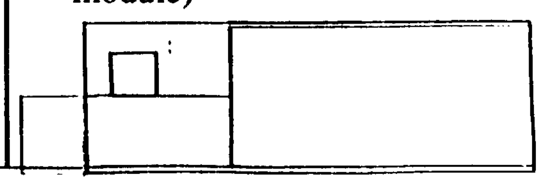
Assembly Mission Two

- MEV Ascent Stage
- MTV Hab Module
- Airlock and tunnel for MTV Hab
- MCRV



Assembly Mission Three

- MEV Descent Stage (includes 2 rovers)
- Surface Payload Module
- Airlock for Surface Payload Module (integrated with surface module)



NTR - Manifesting and Packaging (continued)

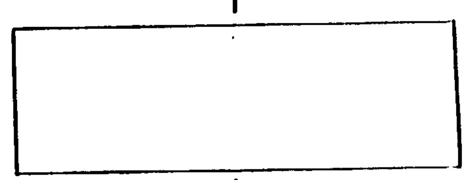
(10m x 30m Shroud, 140 mt HLLV using I-Beam Platform)

Assembly Mission Four

- MEV Aeroshell (5 Sections)

Assembly Mission Five

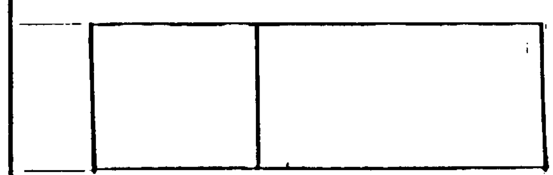
- MEV Aeroshell (5 Sections)



Assembly Mission Six

- In-Line Tank + Fuel
- Tank-to-Truss Structure
- Tank-to-Shield Structure
- Engine/Shield

(These components are integrated as a single package

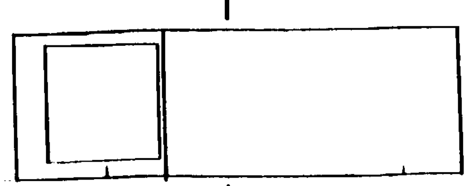


Assembly Mission Seven

- Mars Orbit Capture Tank and fuel (1 of 2)
- Off-loaded fuel tanker (1 of 2)

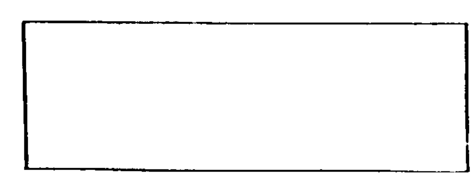
Assembly Mission Eight

- Mars Orbit Capture Tank and fuel (2 of 2)
- Off-loaded fuel tanker (2 of 2)



Assembly Mission Nine and Ten

- TMI Tanks + Fuel (2)



NTR - Manifesting and Packaging

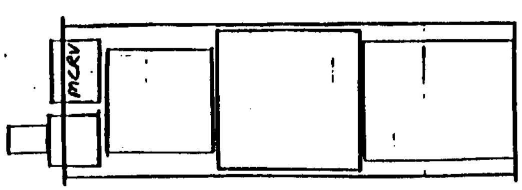
(10m x 30m Shroud, 140 mt HLLV using "Smart" HLLV Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) with 140 metric ton capability and 10m x 30m shroud
- Sequencing based on "smart" HLLV Assembly Platform concept
- Only first HLLV is "smart"; shroud is not shed in order to provide partial debris protection
- "Smart" HLLV contains all necessary data, communications, rcs, reboost equipment; occasional refueling may be provided by SSF based CTV
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NTR configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assembled on orbit (in ten pieces) and requires two dedicated HLLV flights
- 50 mt of propellant off-loaded from TMI tanks, are carried up in tankers along with MOC tanks missions four and five

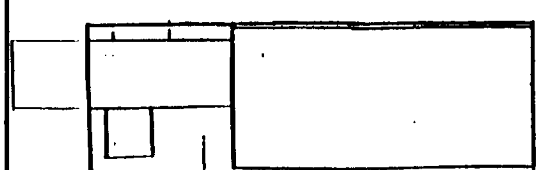
Assembly Mission One

- NTR Truss
- MTV Hab Module
- MEV Ascent stage
- MTV-to-MEV Airlock and Tunnel
- MCRV



Assembly Mission Two

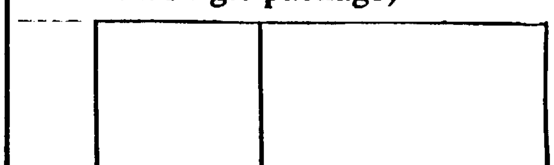
- MEV Descent Stage
- Surface Payload Module
- Airlock for Surface Payload Module (integrated with surface module)



Assembly Mission Three

- In-Line Tank + Fuel
- Tank-to-Truss Structure
- Tank-to-Shield Structure
- Engine/Shield

(these components are integrated as a single package)



NTR - Manifesting and Packaging (continued)

(10m x 30m Shroud, 140 mt HLLV using "Smart" HLLV Platform)

ADVANCED CIVIL SPACE SYSTEMS

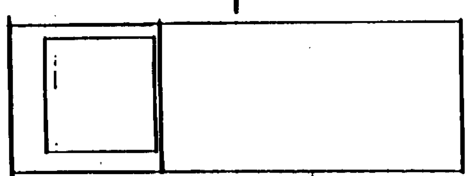
BOEING

Assembly Mission Four

- Mars Orbit Capture Tank and fuel (1 of 2)
- Off-loaded fuel tanker (1 of 2)

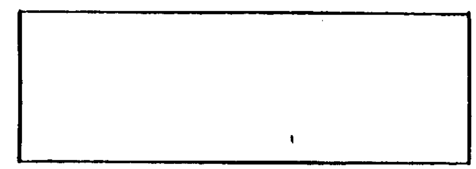
Assembly Mission Five

- Mars Orbit Capture Tank and fuel (2 of 2)
- Off-loaded fuel tanker (2 of 2)



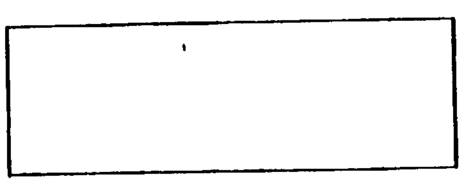
Assembly Mission Six

- TMI Tank + Fuel (1 of 2)



Assembly Mission Seven

- TMI Tank + Fuel (2 of 2)

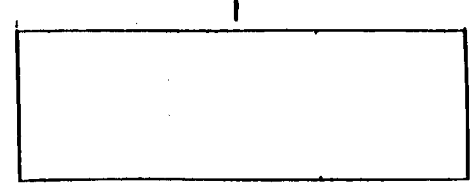


Assembly Mission Eight

- MEV Aeroshell (5 Sections)

Assembly Mission Nine

- MEV Aeroshell (5 Sections)



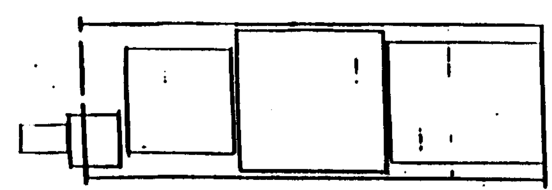
NTR - Manifesting and Packaging
(Mixed HLLV Fleet, using "Smart" HLLV Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) mixed fleet consists of:
  - HLLV #1: 84 metric ton payload capability with 10m x 30m shroud
  - HLLV #2: 120 metric ton payload capability with 7.6m x 30m shroud
- Sequencing based on "smart" HLLV Assembly Platform concept
- Only first HLLV is "smart"; it contains all necessary communications, data, rcs, GNC equipment
- Occasional refueling of reboost system may be necessary and may be accomplished by SSF based CTV
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NTR configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assumed to be integrated at launch ("Ninja Turtle" concept) with other payload packaged in shroud
- Total of 203 tons of propellant, off-loaded from all tanks, are carried up in two tankers in fueling missions eight and nine

Assembly Mission One (HLLV #1)

- NTR Main Truss
- MEV Ascent System
- MTV Hab Module
- MTV-to-MEV Airlock and Tunnel

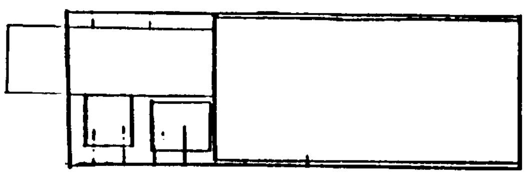


NTR - Manifesting and Packaging (continued)

(Mixed HLLV Fleet, using "Smart" HLLV Platform)

Assembly Mission Two (HLLV #1)

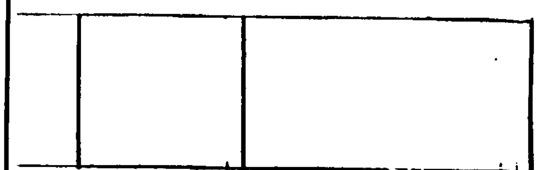
- MEV Descent Stage (includes 2 rovers)
- Mars Surface Payload Module
- Mars Surface Module Airlock (integrated with surface module)
- MCRV
- Aerobrake



Assembly Mission Three (HLLV #1)

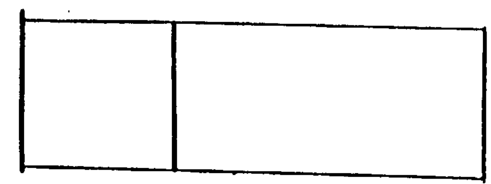
- In-Line Tank + Fuel
- Tank-to-Truss Structure
- Tank-to-Shield Structure
- Engine/Shield

(These components are integrated as a single package)



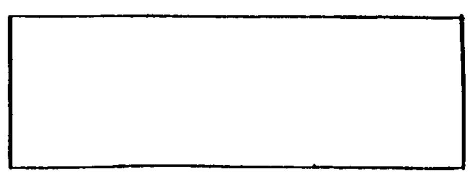
Assembly Mission Four and Five (HLLV #1)

- Mars Orbit Capture Tanks + Fuel (2)



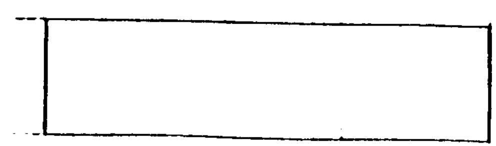
Assembly Mission Six and Seven (HLLV #1)

- TMI Tanks + Fuel (2)



Assembly Mission Eight and Nine (HLLV #2)

- Off-Loaded fuel Tankers and total of 203 tons of fuel



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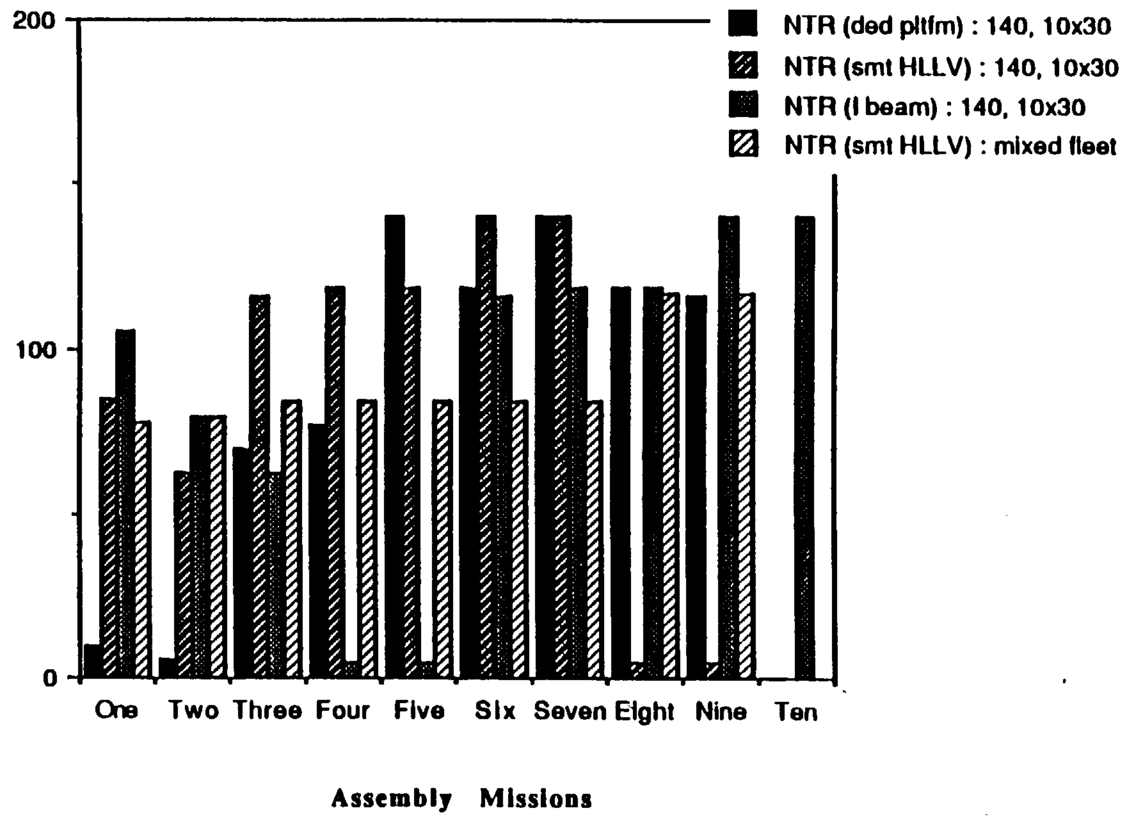
NTR Manifesting and Packaging

Comparison of Manifest Data for NTR Vehicle for Several On-Orbit Platform Concepts



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NTR On-Orbit Assembly Assumptions Using Dedicated Assembly Platform

- NTR main truss is self-deploying.

- Aerobrake assembly does not commence until NTR main truss is deployed and secured to Assembly Platform.

- All assembly robotically performed with EVA contigency.

- CTV capable of maneuvering 140 metric tons.

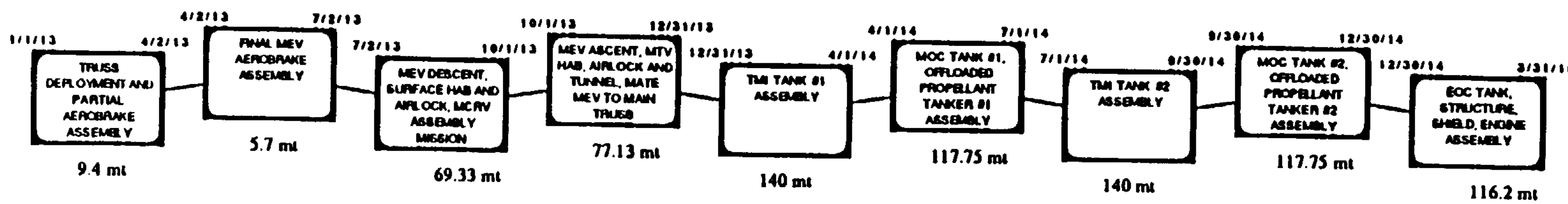
- PRMS can manipulate 140 metric tons.

- Assembly platform has a pressurized "control station" that can locally control robotic activity and provide life support for crew during assembly operations.

- Pieces of the vehicle can be securely stored on the assembly platform and capability exists to shift mass along the truss work of the platform.

- Platform is equipped with communications, data, reboost, attitude control equipment and debris shielding where needed.

NTR Top Level Assembly Using Dedicated Platform

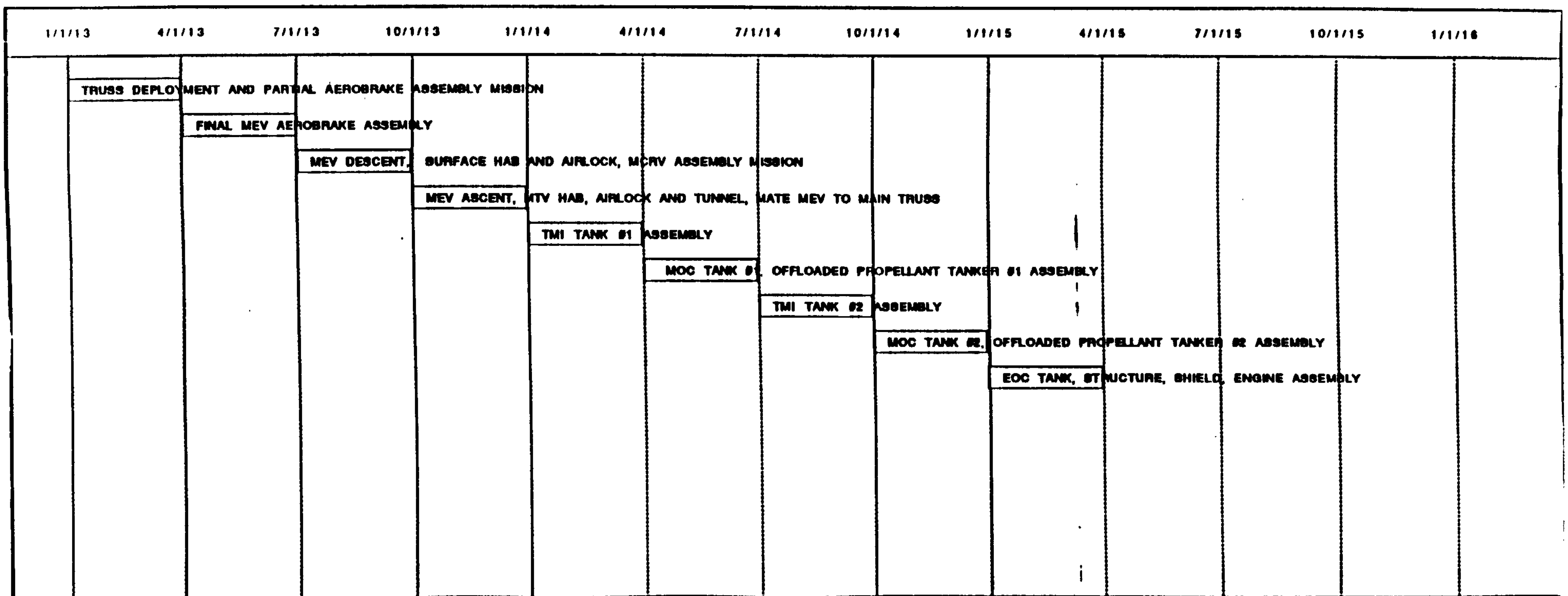


- Smallest unit of time is 1 hour
- 16 hours = 1 day of Assembly Duration

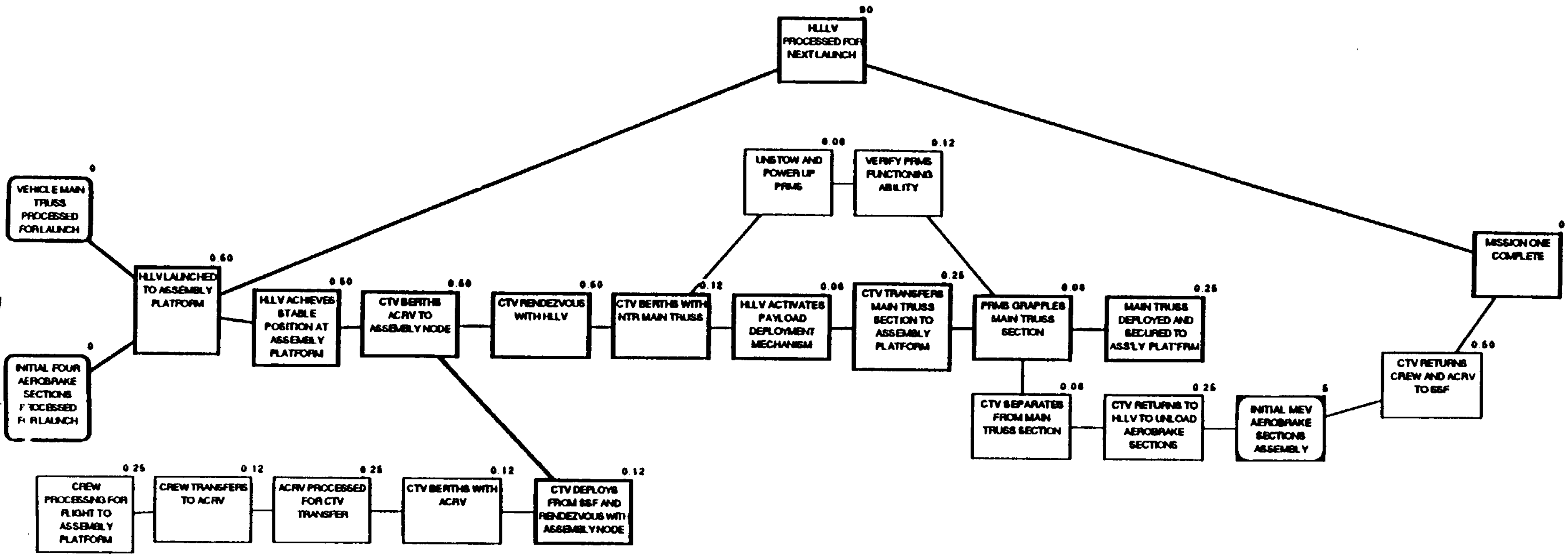
BASELINE DURATIONS:
- HLLV Launch = .5 day
- HLLV achieves stable orbit = .25 day
- MUV deploys from/to Freedom = .5 day
- MUV berths to components = .25 day
- Unstow and power up Robotics = .06 day
- Robotic verification = .12 day
- HLLV deploys components = .06 day
- MUV transfers components = .25 day
- Robotic tasks = .06 day
- EVA/Robotic Contingency = .5 day
- Component Inspection = .12 day
- Component Test = .25 day
- Subassemblies to stand-by mode = .5 day
- Mechanical Fastening of components = .18 day
- Crew processing for flight = .25 day

NTR Top Level Assembly Schedule

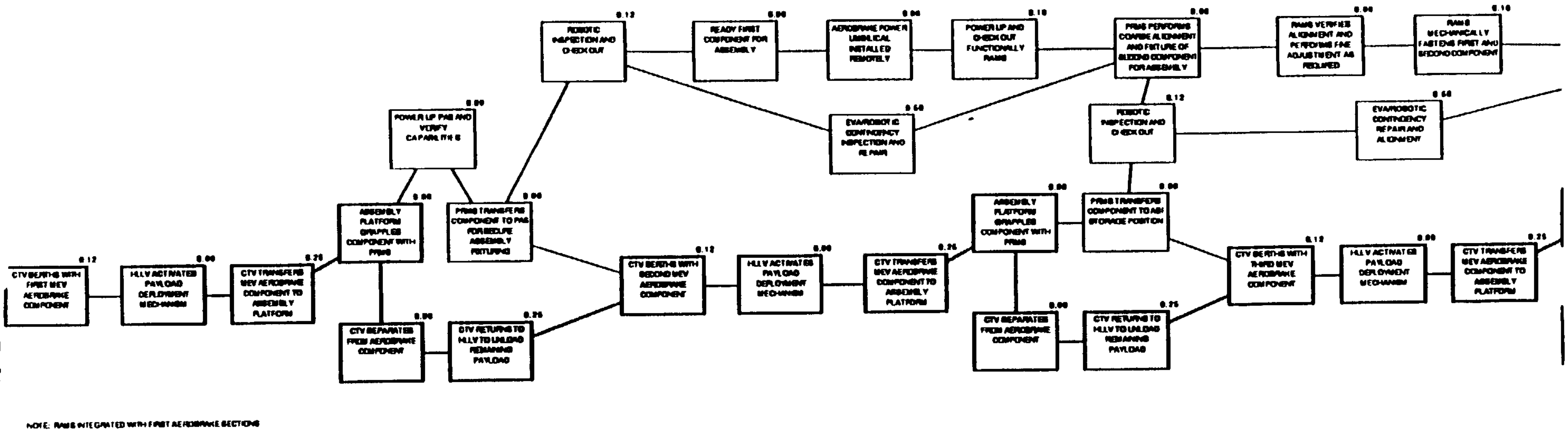
Name	Earliest Start	Earliest Finish	Subproject	Days
TRUSS DEPLOYMENT AND PARTIAL AEROBRAKE ASSEMBLY MISSION	1/1/13	4/2/13	NTR MISSION ONE	91
FINAL MEV AEROBRAKE ASSEMBLY	4/2/13	7/2/13	NTR MISSION TWO	91
MEV DESCENT, SURFACE HAB AND AIRLOCK, MCRV ASSEMBLY MISSION	7/2/13	10/1/13	NTR MISSION THREE	91
MEV ASCENT, MTV HAB, AIRLOCK AND TUNNEL, MATE MEV TO MAIN	10/1/13	12/31/13	NTR MISSION FOUR	91
TMI TANK #1 ASSEMBLY.	12/31/13	4/1/14	NTR MISSION FIVE	91
MOC TANK #1, OFFLOADED PROPELLANT TANKER #1 ASSEMBLY	4/1/14	7/1/14	NTR MISSION SIX	91
TMI TANK #2 ASSEMBLY	7/1/14	9/30/14	NTR MISSION SEVEN	91
MOC TANK #2, OFFLOADED PROPELLANT TANKER #2 ASSEMBLY	9/30/14	12/30/14	NTR MISSION EIGHT	91
EOC TANK, STRUCTURE, SHIELD, ENGINE ASSEMBLY	12/30/14	3/31/15	NTR MISSION NINE	91

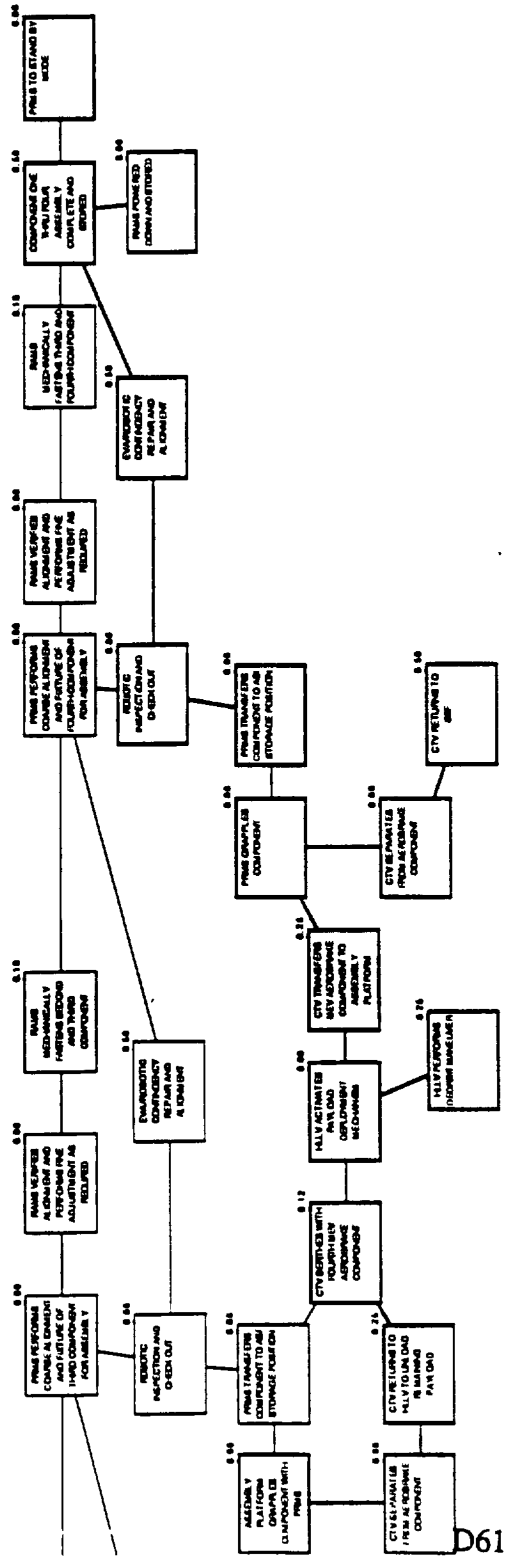


NTR MISSION ONE

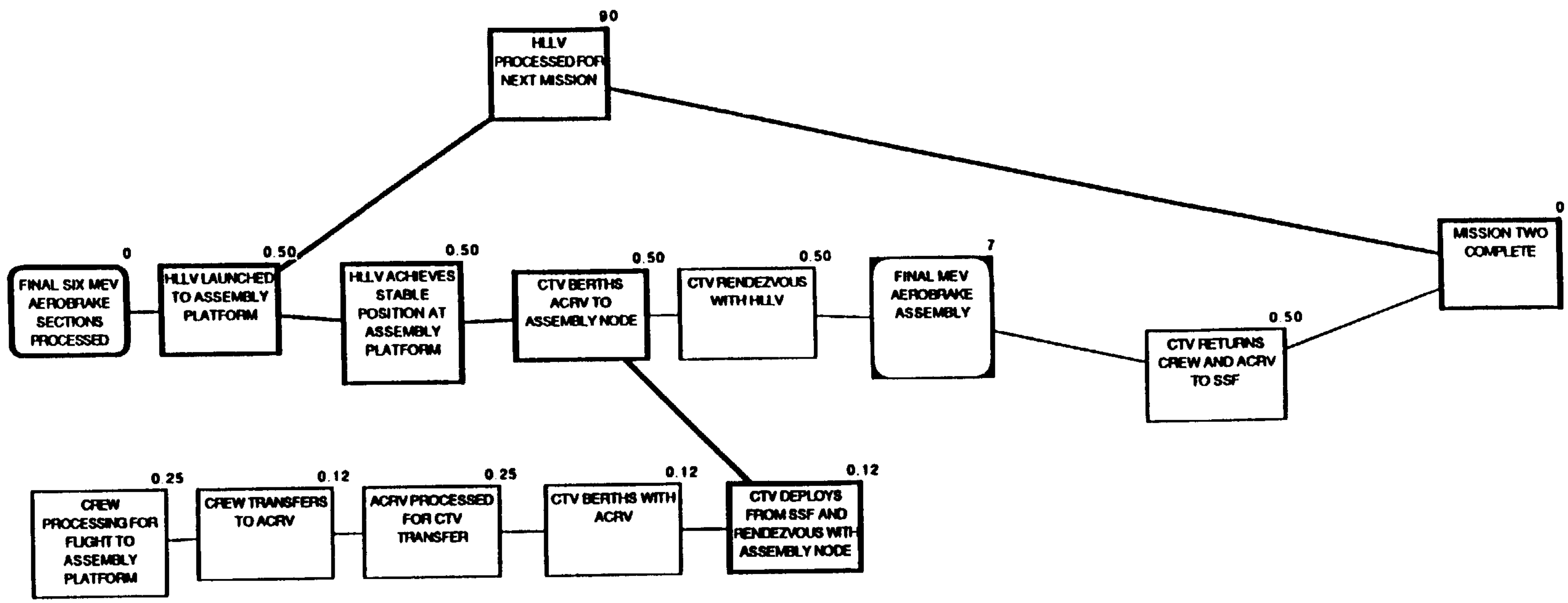


MEV AEROBRAKE MISSION ONE

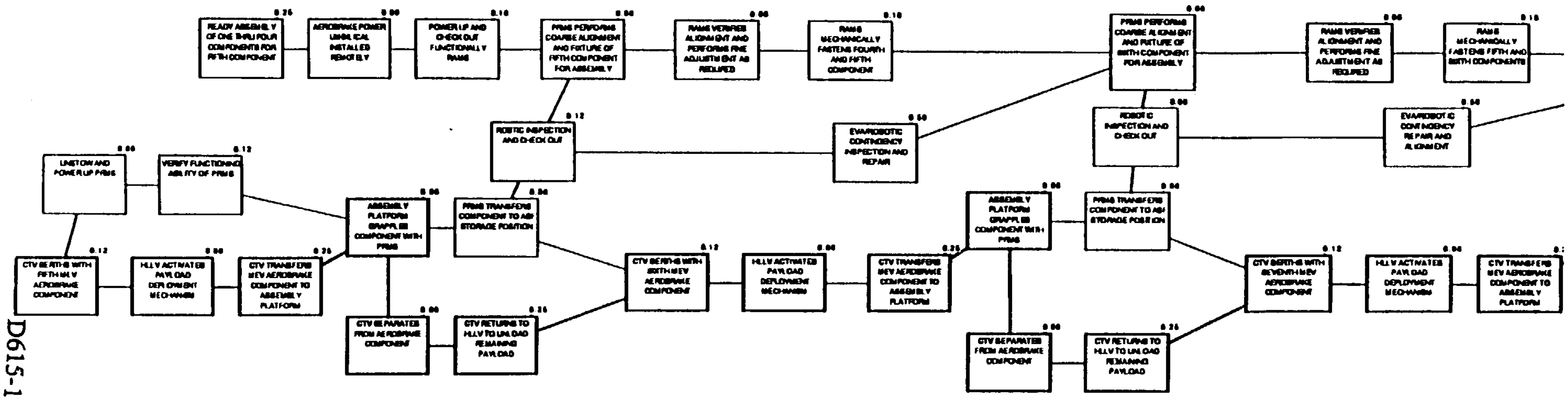




NTR MISSION TWO



MEV AEROBRAKE MISSION TWO NTR



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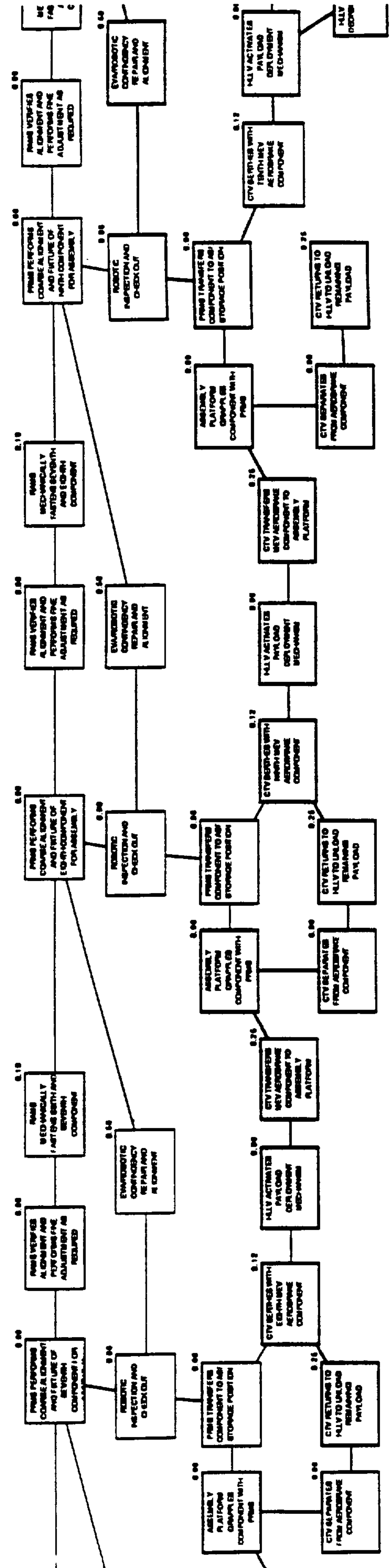
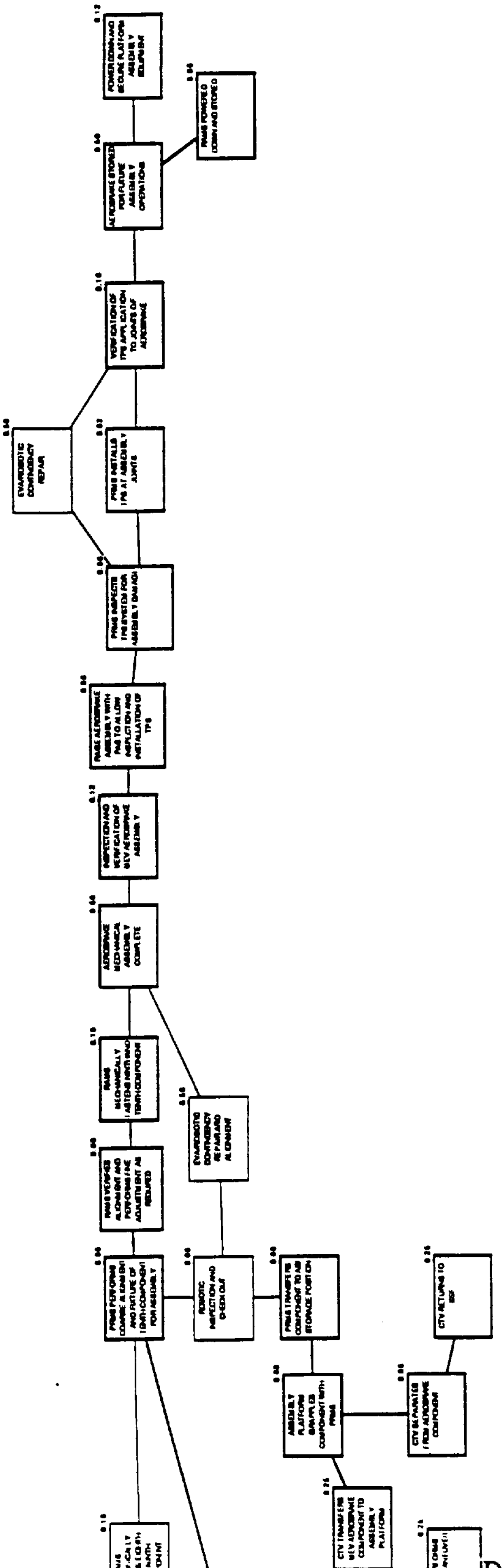
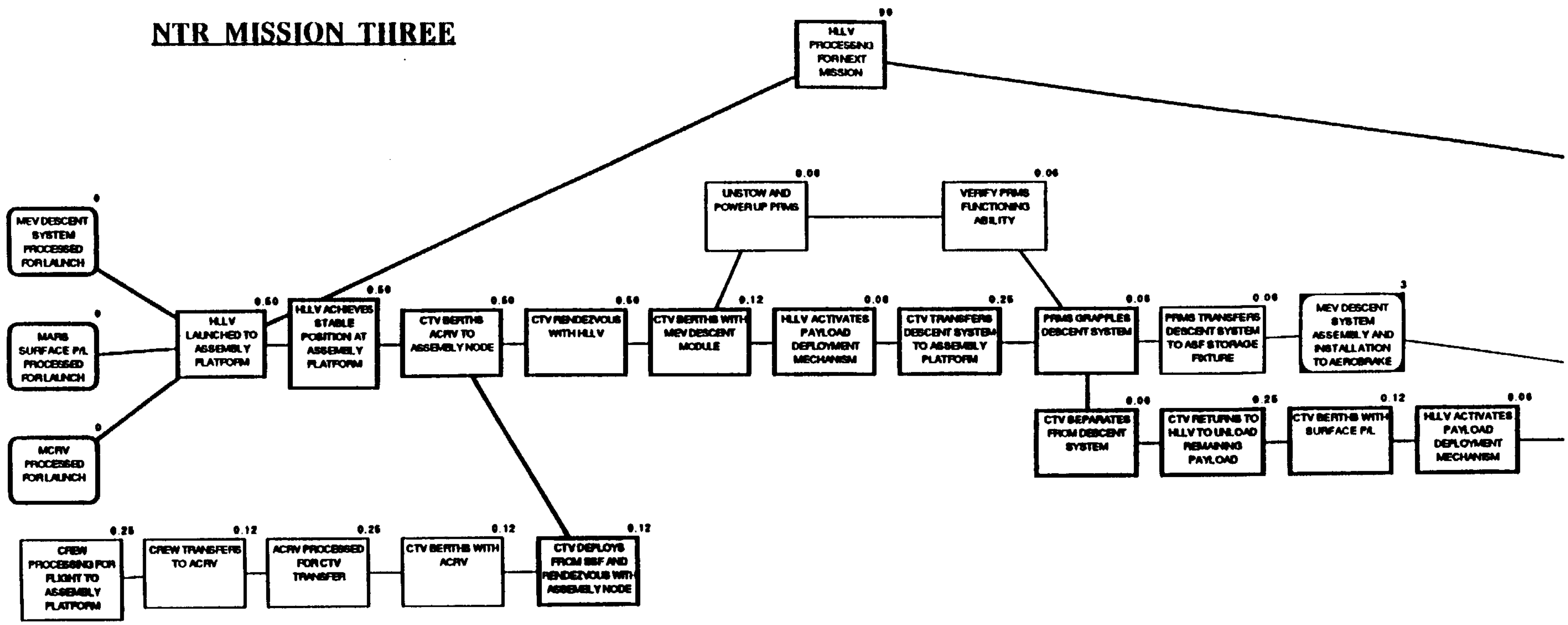
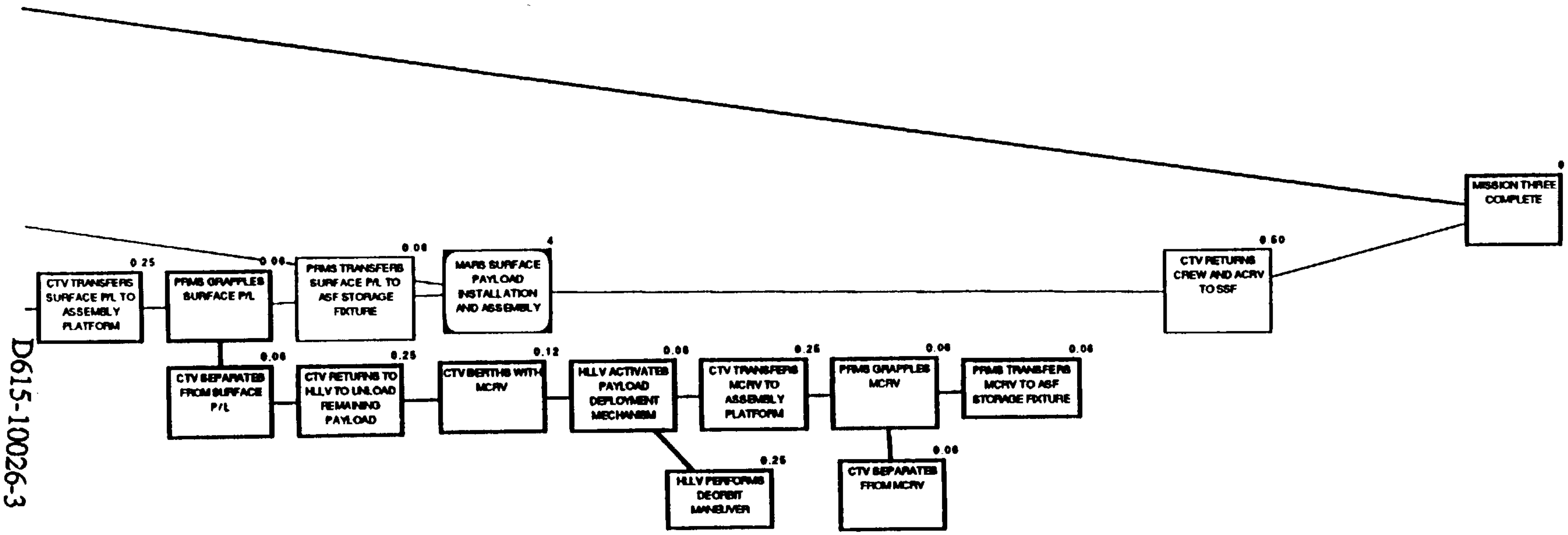


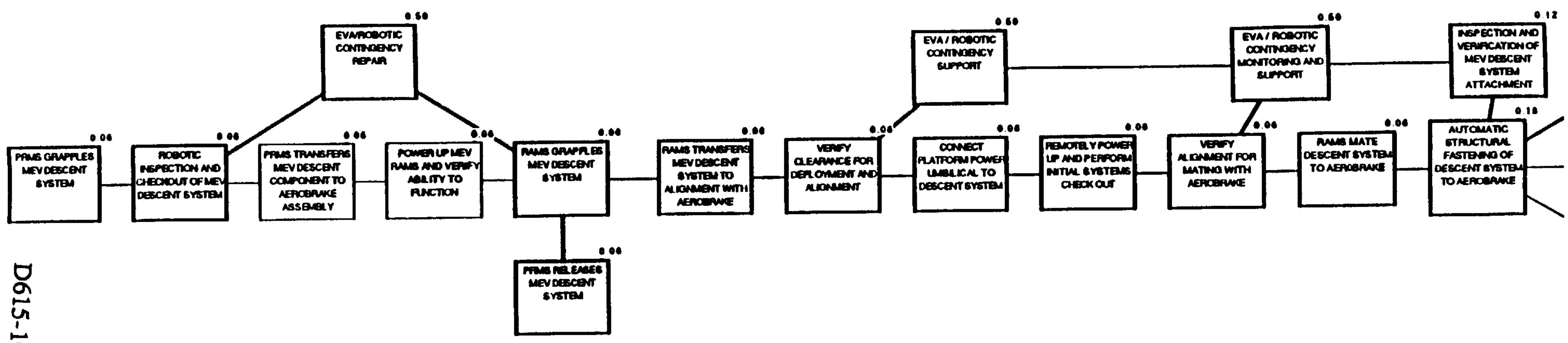

POWER DOWN AND SECURE PLATFORM ASSEMBLY EQUIPMENT
AEROBRAKE STORED FOR FUTURE ASSEMBLY OPERATIONS
RAMS POWERED DOWN AND STORED
VERIFICATION OF TPS APPLICATION TO JOINTS OF AEROBRAKE
EVA/ROBOTIC CONTINGENCY REPAIR
PRMS INSTALLS TPS AT ASSEMBLY JOINTS
PRMS INSPECTS TPS SYSTEM FOR ASSEMBLY DAMAGE
RAISE AEROBRAKE ASSEMBLY WITH PAS TO ALLOW INSPECTION AND INSTALLATION OF TPS
INSPECTION AND VERIFICATION OF MEV AEROBRAKE ASSEMBLY
AEROBRAKE MECHANICAL ASSEMBLY COMPLETE
RAMS MECHANICALLY FASTENS NINTH AND TENTH COMPONENT
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF TENTH COMPONENT FOR ASSEMBLY
ROBOTIC INSPECTION AND CHECK OUT
PRMS TRANSFERS COMPONENT TO AB STORAGE POSITION
CTV RETURNS TO SSF
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
CTV SEPARATES FROM AEROBRAKE COMPONENT
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM

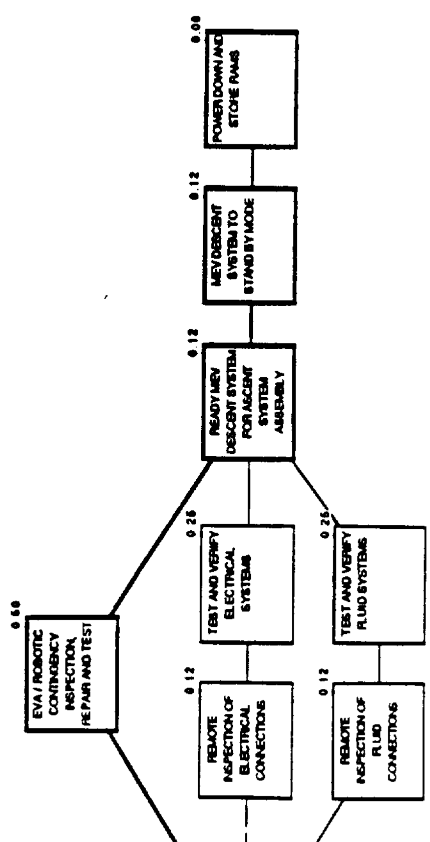
NTR MISSION THREE



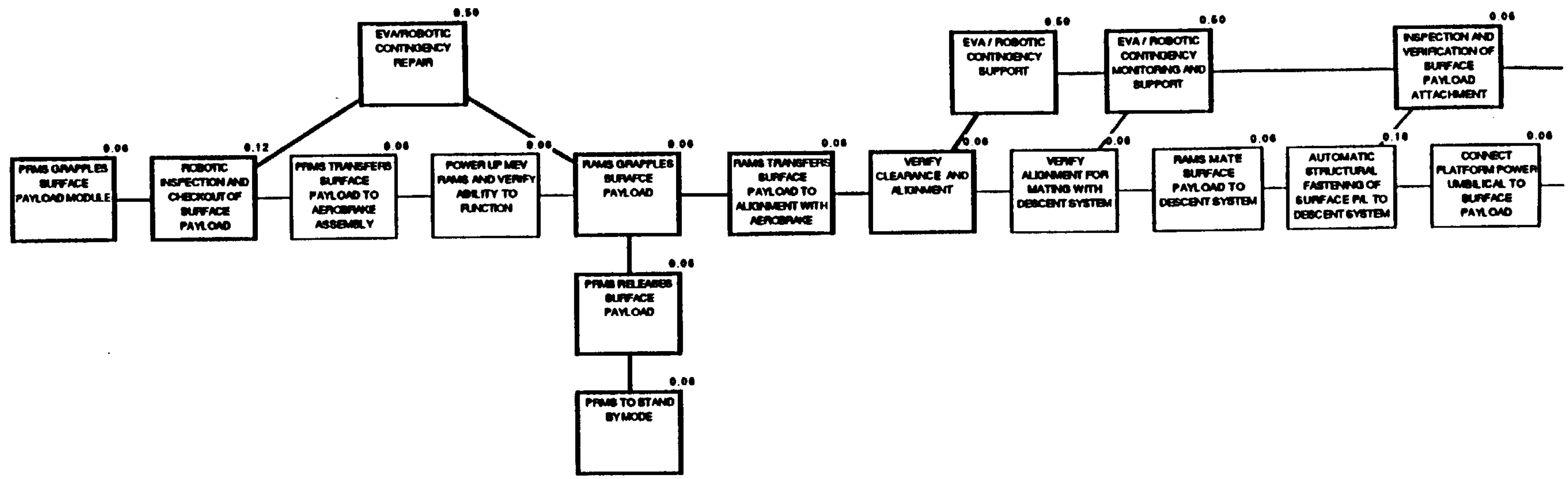

0.25
CTV TRANSFERS SURFACE P/L TO ASSEMBLY PLATFORM
0.06
PRMS GRAPPLES SURFACE P/L
0.06
PRMS TRANSFERS SURFACE P/L TO ASF STORAGE FIXTURE
4
MARS SURFACE PAYLOAD INSTALLATION AND ASSEMBLY
0.50
CTV RETURNS CREW AND ACRV TO SSF
0
MISSION THREE COMPLETE
0.06
CTV SEPARATES FROM SURFACE P/L
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.12
CTV BERTHS WITH MCRV
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.25
CTV TRANSFERS MCRV TO ASSEMBLY PLATFORM
0.06
PRMS GRAPPLES MCRV
0.06
CTV SEPARATES FROM MCRV
0.06
PRMS TRANSFERS MCRV TO ASF STORAGE FIXTURE

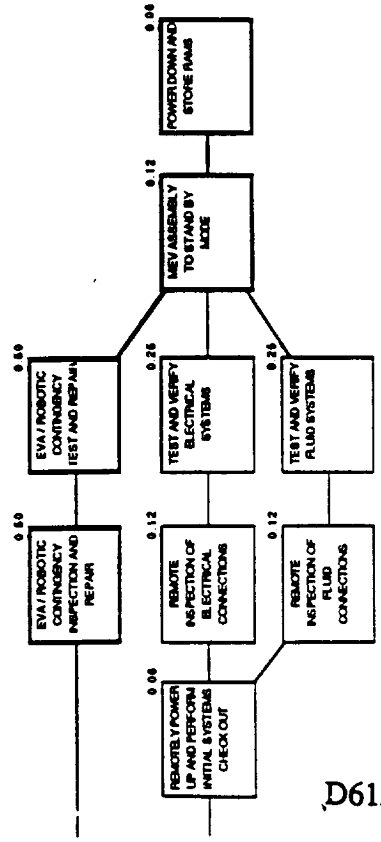
MEV DESCENT SYSTEM NTR



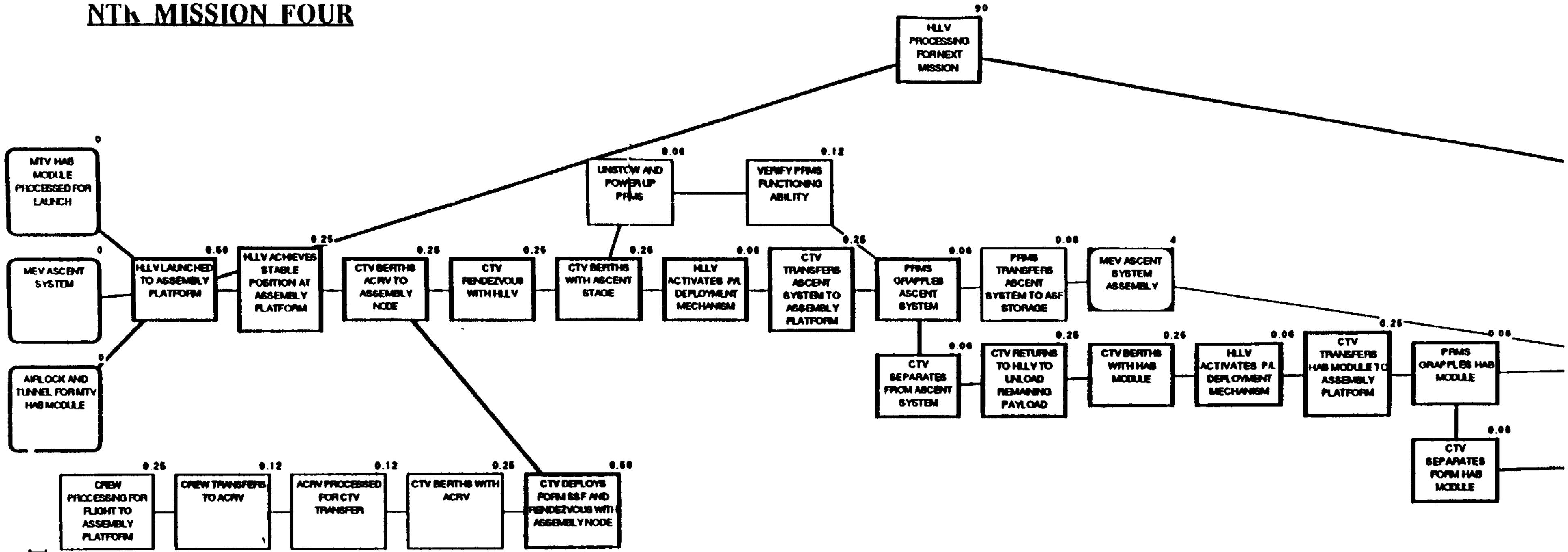

0.50
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
0.12
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
READY MEV DESCENT SYSTEM FOR ASCENT SYSTEM ASSEMBLY
0.12
MEV DESCENT SYSTEM TO STAND BY MODE
0.08
POWER DOWN AND STORE RAMS

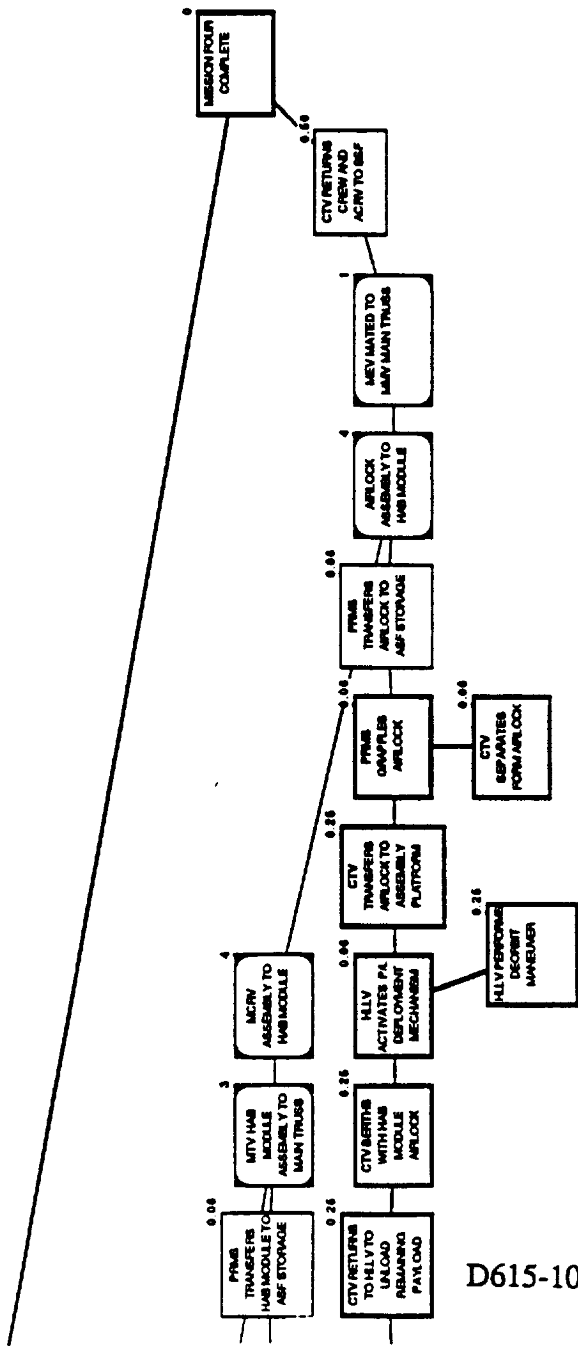
MARS SURFACE PAYLOAD NTR



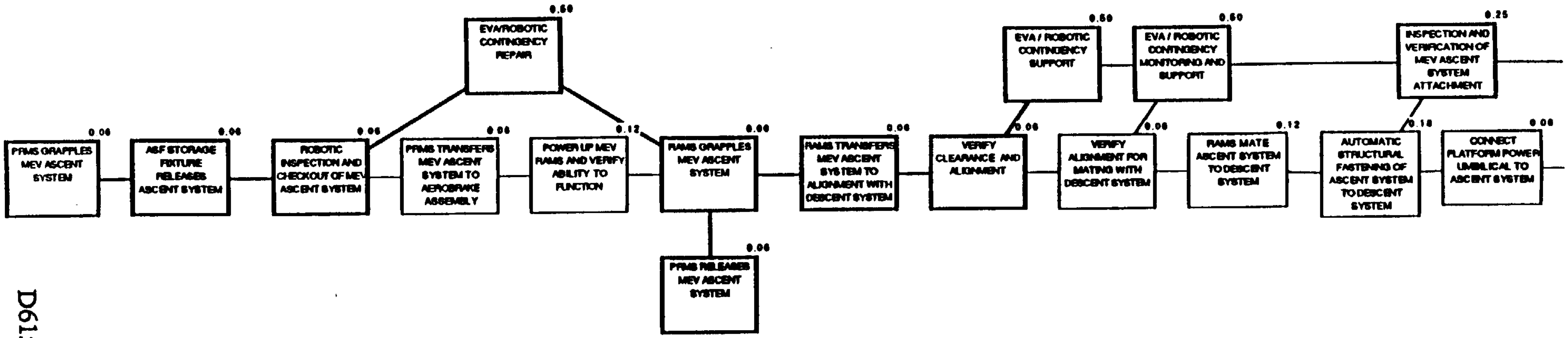

0.06
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.12
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
MEV ASSEMBLY TO STAND BY MODE
0.06
POWER DOWN AND STORE RAMS

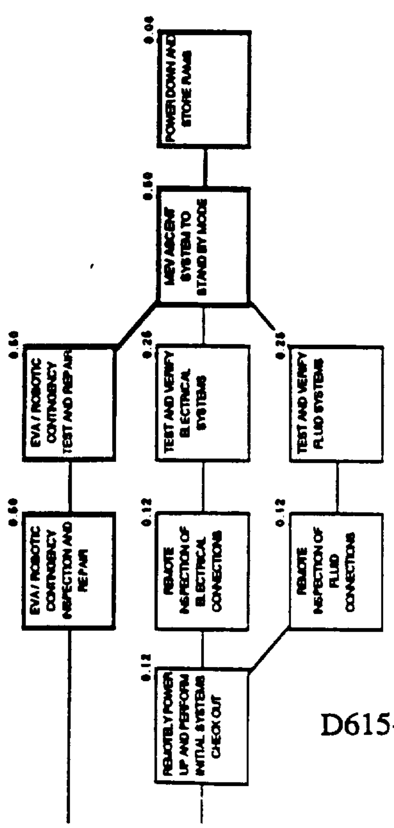
NTR MISSION FOUR



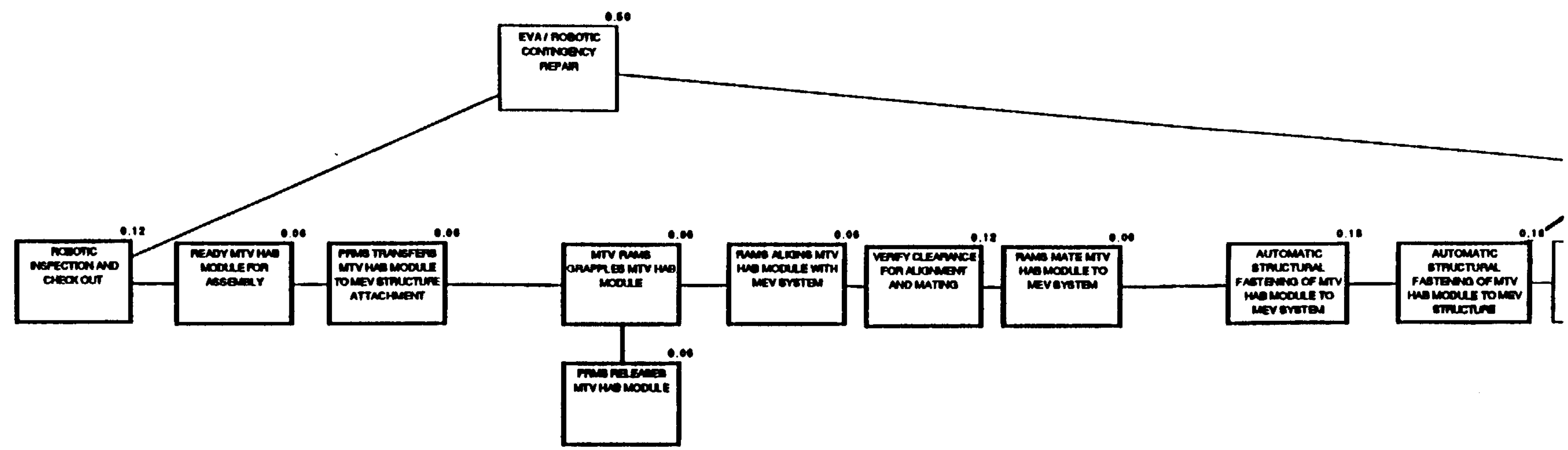

0
MISSION FOUR COMPLETE
0.50
CTV RETURNS CREW AND ACRV TO SSF
1
MEV MATED TO MTV MAIN TRUSS
4
AIRLOCK ASSEMBLY TO HAB MODULE
0.06
PRMS TRANSFERS AIRLOCK TO ASF STORAGE
0.06
PRMS GRAPPLES AIRLOCK
0.06
CTV SEPARATES FORM AIRLOCK
0.25
CTV TRANSFERS AIRLOCK TO ASSEMBLY PLATFORM
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.06
HLLV ACTIVATES P/L DEPLOYMENT MECHANISM
4
MCRV ASSEMBLY TO HAB MODULE
3
MTV HAB MODULE ASSEMBLY TO MAIN TRUSS
0.25
CTV BERTHS WITH HAB MODULE AIRLOCK
0.06
PRMS TRANSFERS HAB MODULE TO ASF STORAGE
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD

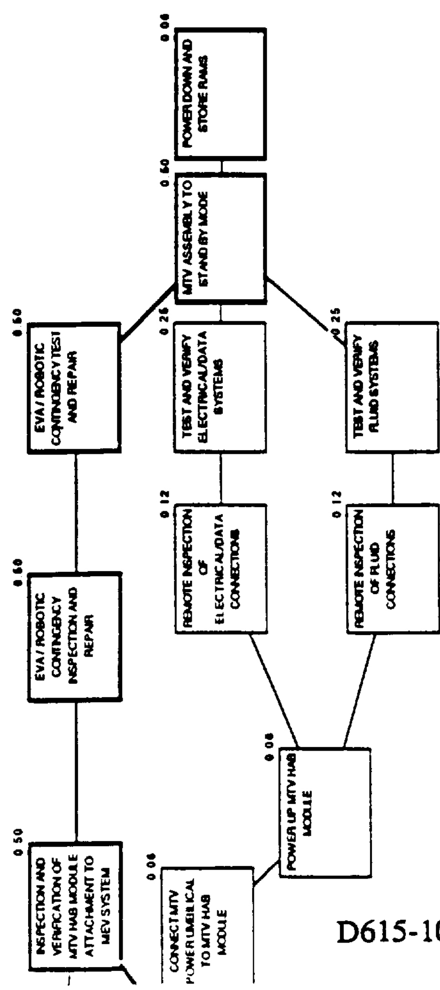
MEV ASCENT SYSTEM NTR



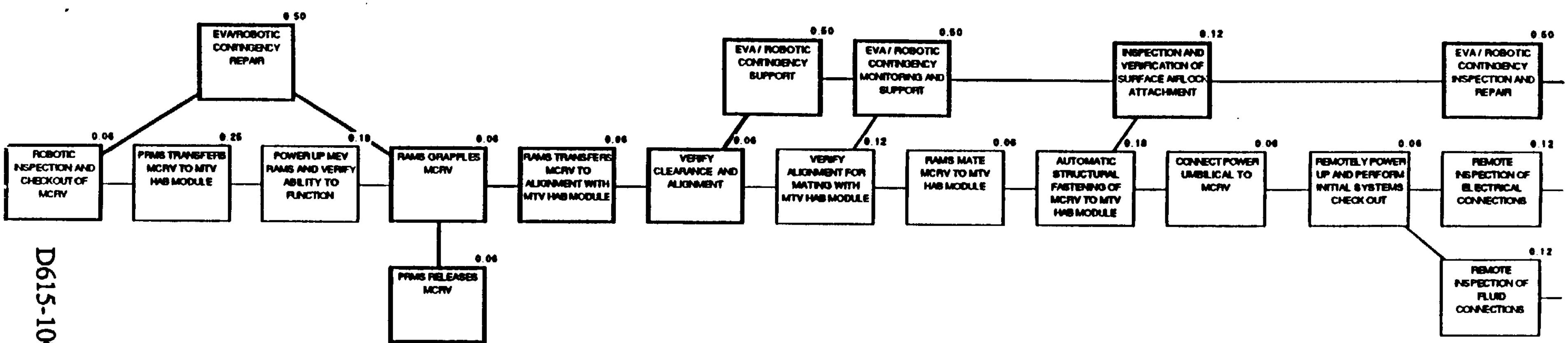

0.12
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.12
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.50
MEV ASCENT SYSTEM TO STAND BY MODE
0.04
POWER DOWN AND STORE RAMS

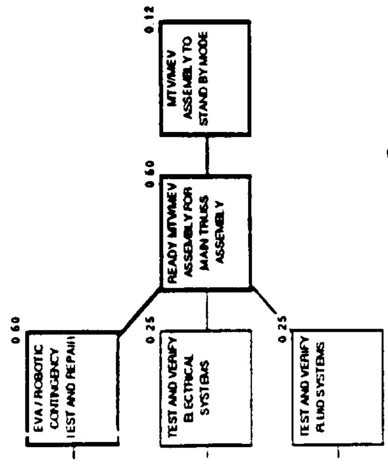
NTR MTV HAB MODULE ASSEMBLY



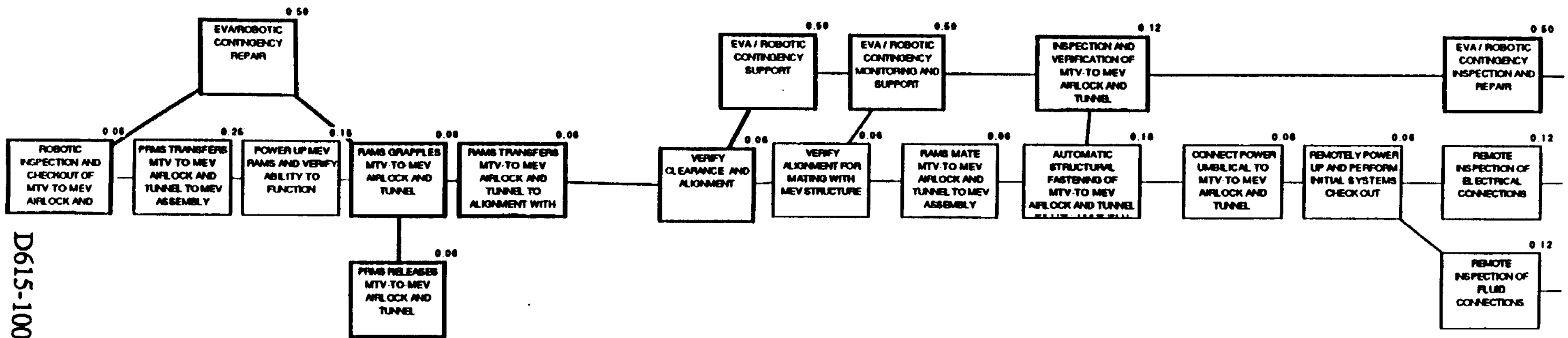

0.50
INSPECTION AND VERIFICATION OF MTV HAB MODULE ATTACHMENT TO MEV SYSTEM
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.50
MTV ASSEMBLY TO STAND BY MODE
0.06
POWER DOWN AND STORE RAMS
0.25
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.06
POWER UP MTV HAB MODULE
0.06
CONNECT MTV POWER UMBILICAL TO MTV HAB MODULE

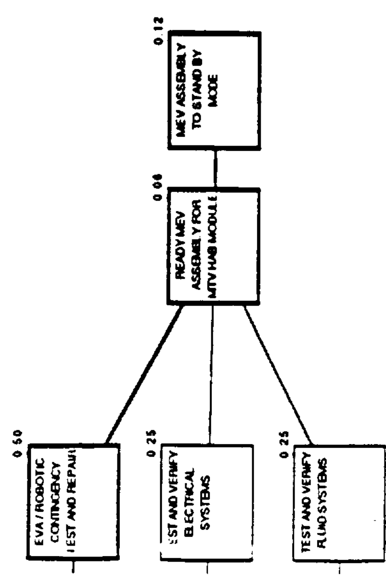
NTR MCRV ASSEMBLY



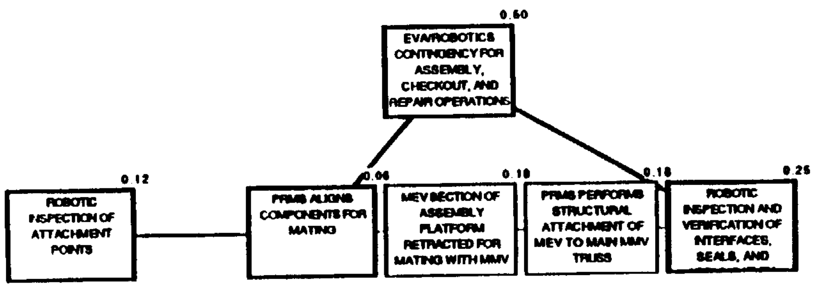

0.12
MTV/MEV ASSEMBLY TO STAND BY MODE
0.60
READY MTV/MEV ASSEMBLY FOR MAIN TRUSS ASSEMBLY
0.60
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.25
TEST AND VERIFY FLUID SYSTEMS

NTR MTV-TO-MEV AIRLOCK AND TUNNEL ASSEMBLY

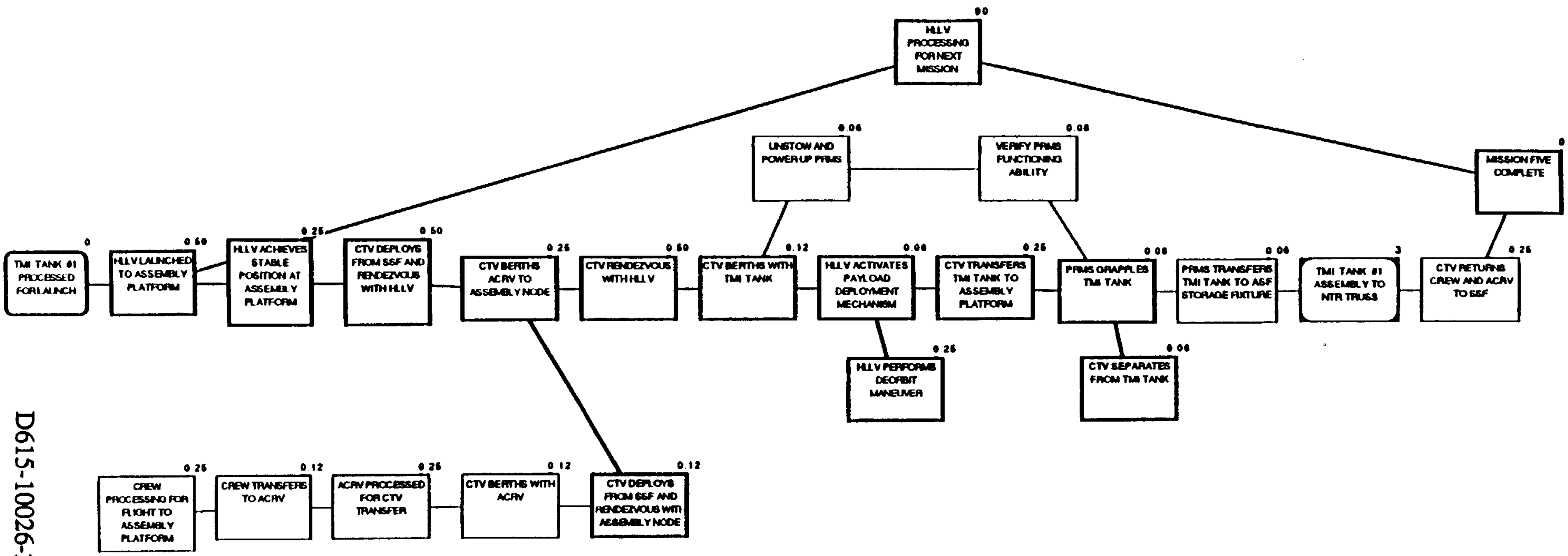



0.12
MEV ASSEMBLY TO STAND BY MODE
0.08
READY MEV ASSEMBLY FOR MTV HAB MODULE
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.25
TEST AND VERIFY FLUID SYSTEMS

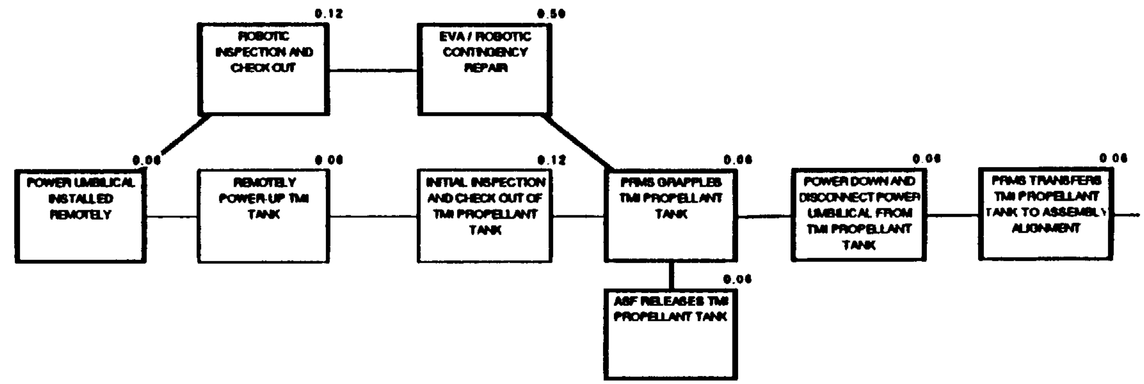
NTR MEV-TO-MTV MATING

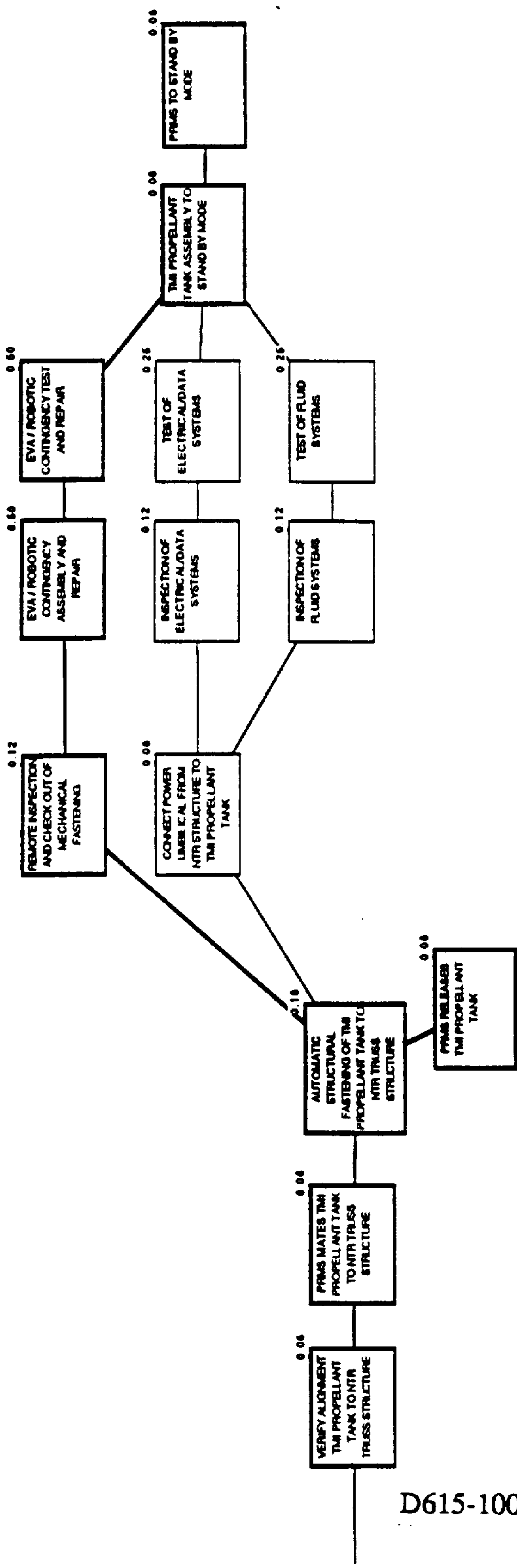


NTR MISSION FIVE

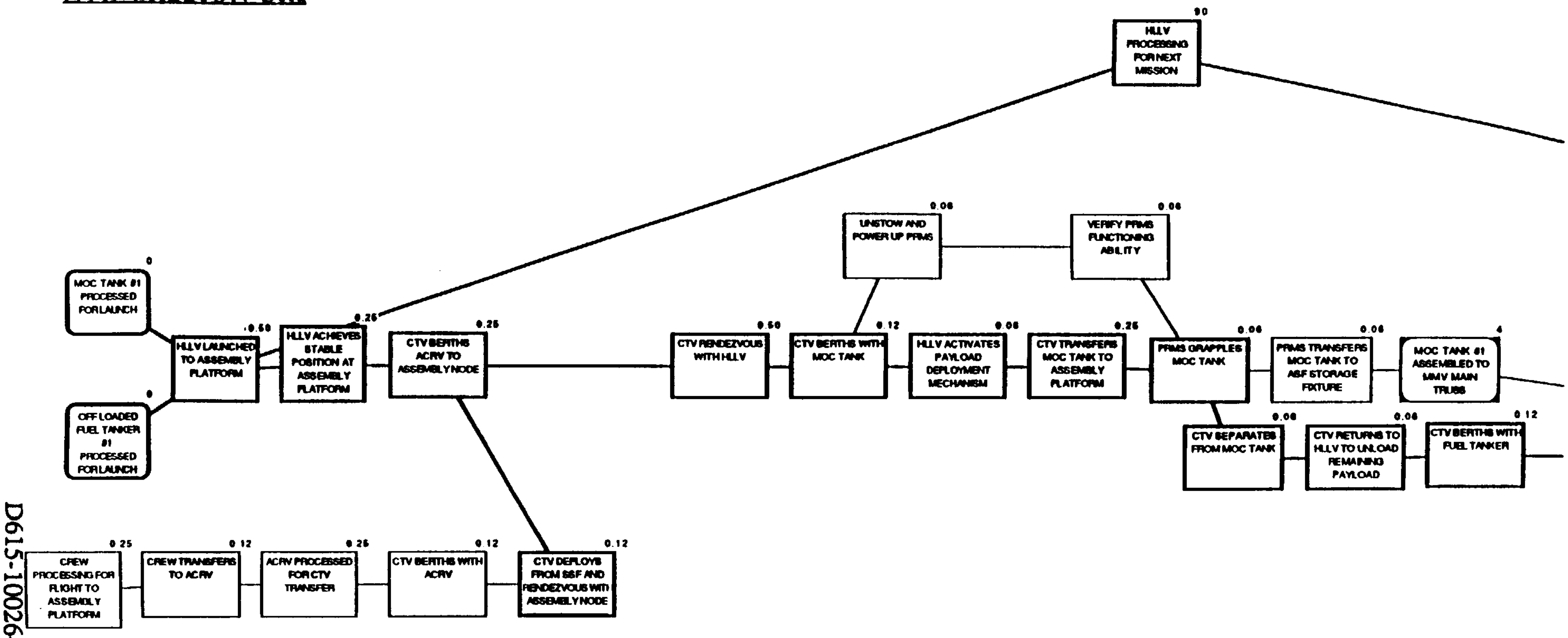


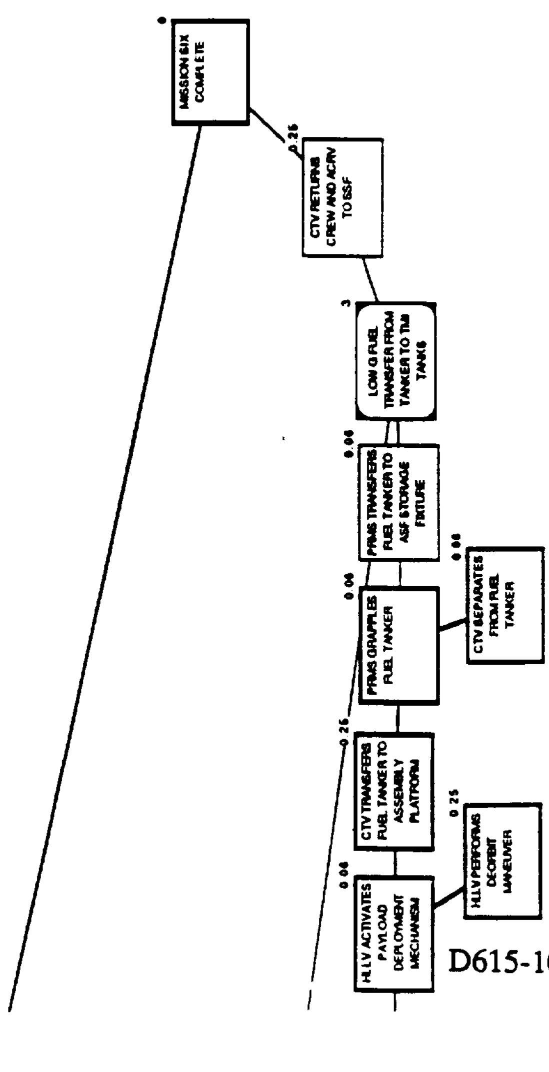
NTR TMI TANK #1 ASSEMBLY



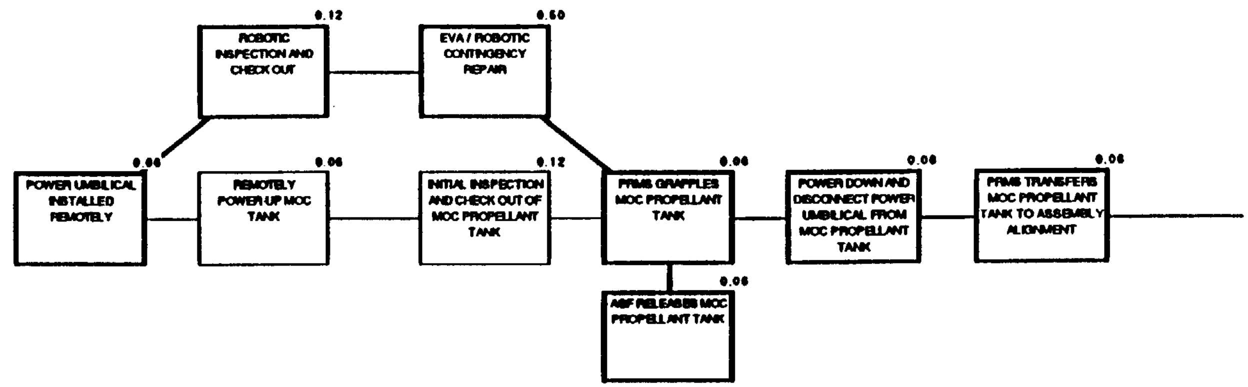

0.06
VERIFY ALIGNMENT TMI PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
PRMS MATES TMI PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.18
AUTOMATIC STRUCTURAL FASTENING OF TMI PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
PRMS RELEASES TMI PROPELLANT TANK
0.12
REMOTE INSPECTION AND CHECK OUT OF MECHANICAL FASTENING
0.06
CONNECT POWER UMBILICAL FROM NTR STRUCTURE TO TMI PROPELLANT TANK
0.50
EVA / ROBOTIC CONTINGENCY ASSEMBLY AND REPAIR
0.12
INSPECTION OF ELECTRICAL/DATA SYSTEMS
0.12
INSPECTION OF FLUID SYSTEMS
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST OF ELECTRICAL/DATA SYSTEMS
0.25
TEST OF FLUID SYSTEMS
0.06
TMI PROPELLANT TANK ASSEMBLY TO STAND BY MODE
0.06
PRMS TO STAND BY MODE

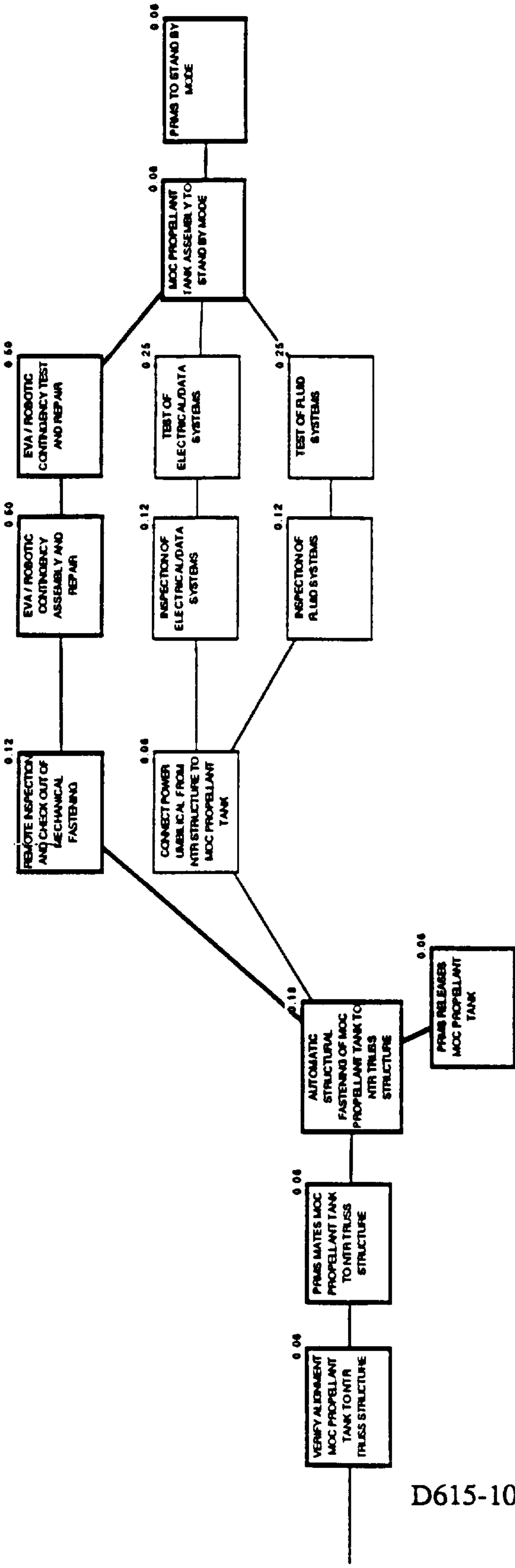
NTR MISSION SIX



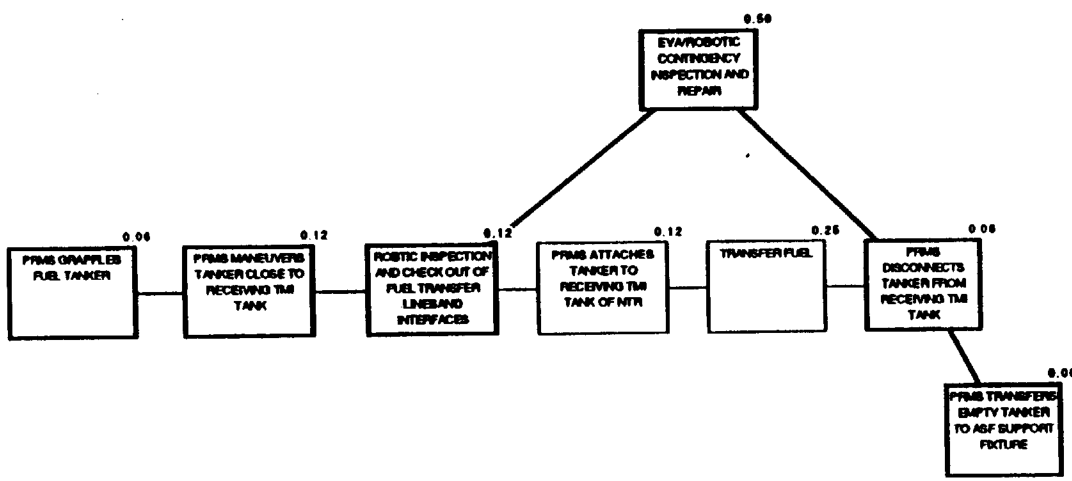

MISSION SIX COMPLETE
0.25
CTV RETURNS CREW AND ACRV TO SSF
3
LOW G FUEL TRANSFER FROM TANKER TO TMI TANKS
0.06
PRMS TRANSFERS FUEL TANKER TO ASF STORAGE FIXTURE
0.06
PRMS GRAPPLES FUEL TANKER
0.06
CTV SEPARATES FROM FUEL TANKER
0.25
CTV TRANSFERS FUEL TANKER TO ASSEMBLY PLATFORM
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.04
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM

NTR MOC PROPELLANT TANK ASSEMBLY

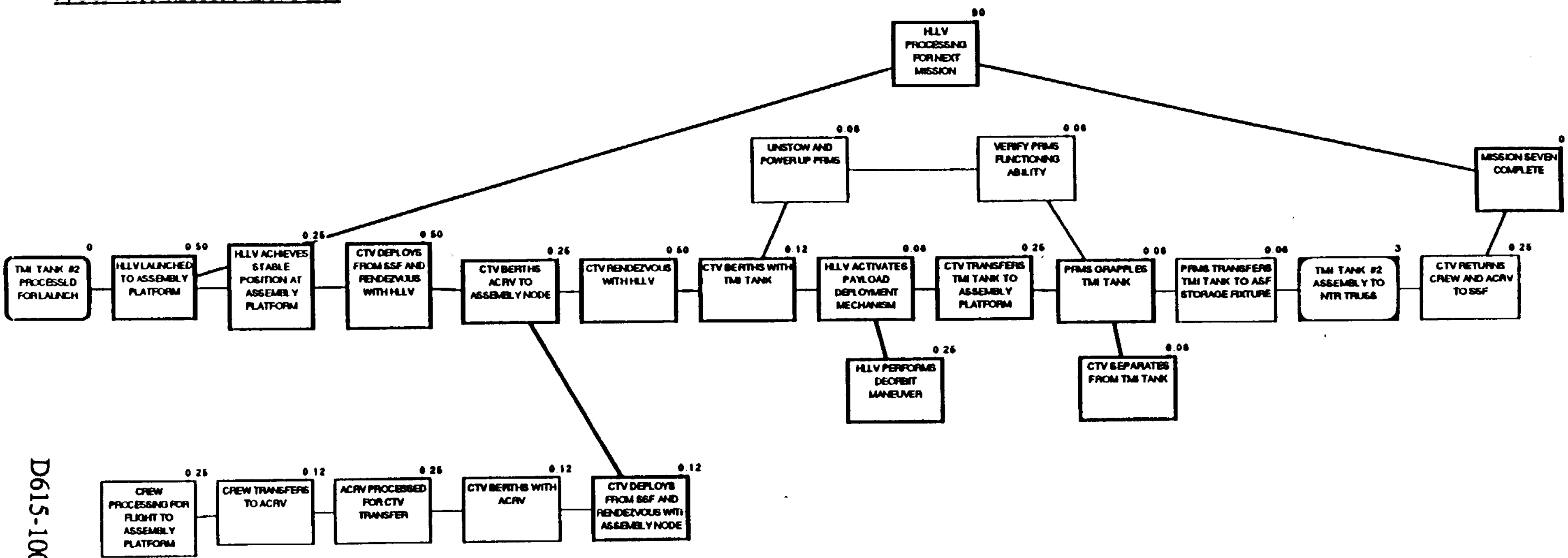



0.06
VERIFY ALIGNMENT MOC PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
PRMS MATES MOC PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.18
AUTOMATIC STRUCTURAL FASTENING OF MOC PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
PRMS RELEASES MOC PROPELLANT TANK
0.12
REMOTE INSPECTION AND CHECK OUT OF MECHANICAL FASTENING
0.06
CONNECT POWER UMBILICAL FROM NTR STRUCTURE TO MOC PROPELLANT TANK
0.50
EVA / ROBOTIC CONTINGENCY ASSEMBLY AND REPAIR
0.12
INSPECTION OF ELECTRICAL/DATA SYSTEMS
0.12
INSPECTION OF FLUID SYSTEMS
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST OF ELECTRICAL/DATA SYSTEMS
0.25
TEST OF FLUID SYSTEMS
0.06
MOC PROPELLANT TANK ASSEMBLY TO STAND BY MODE
0.06
PRMS TO STAND BY MODE

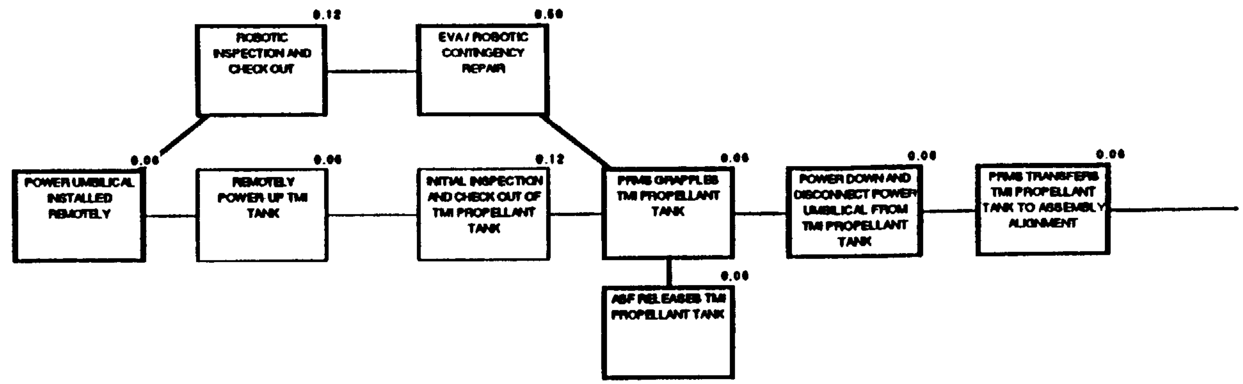
NTR OFFLOADED PROP TRANSFER

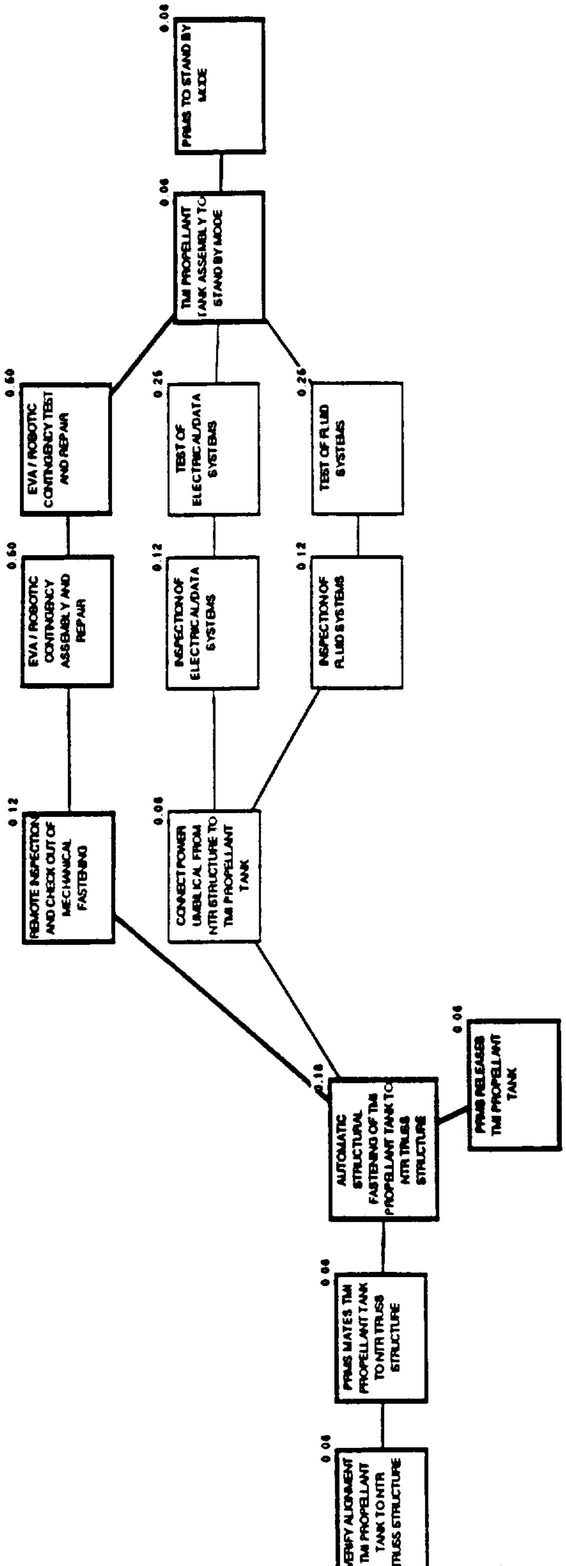


NTR MISSION SEVEN

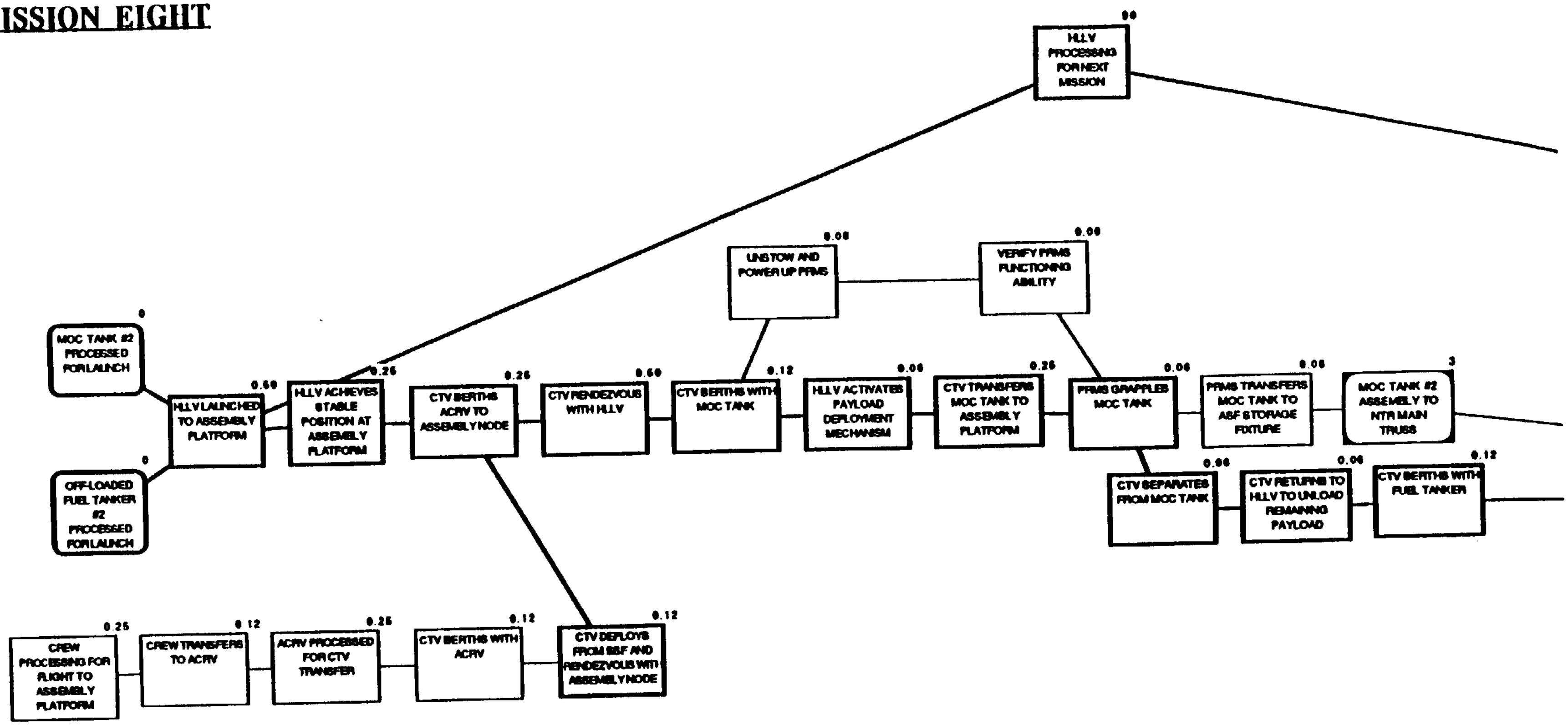


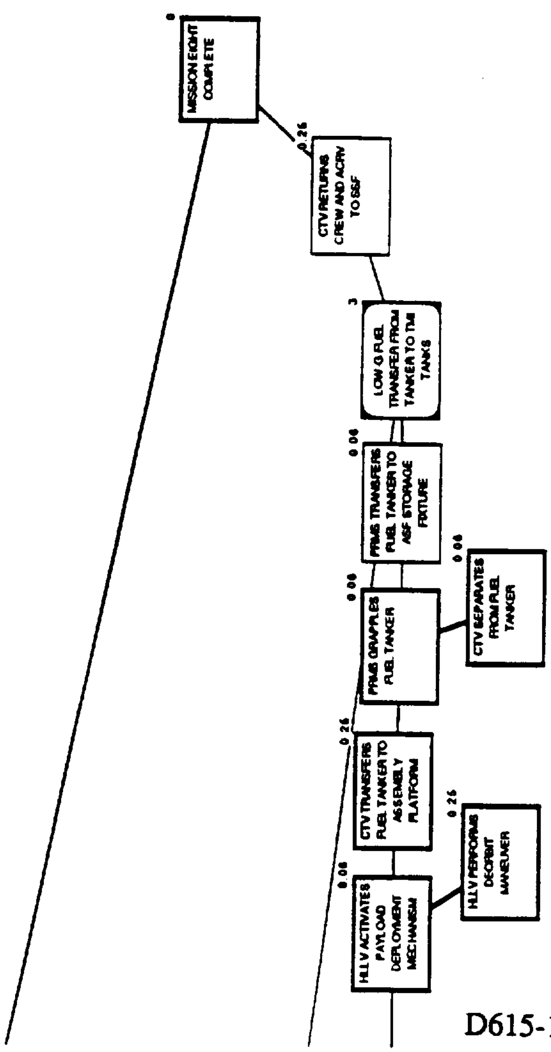
NTR TMI TANK #2 ASSEMBLY



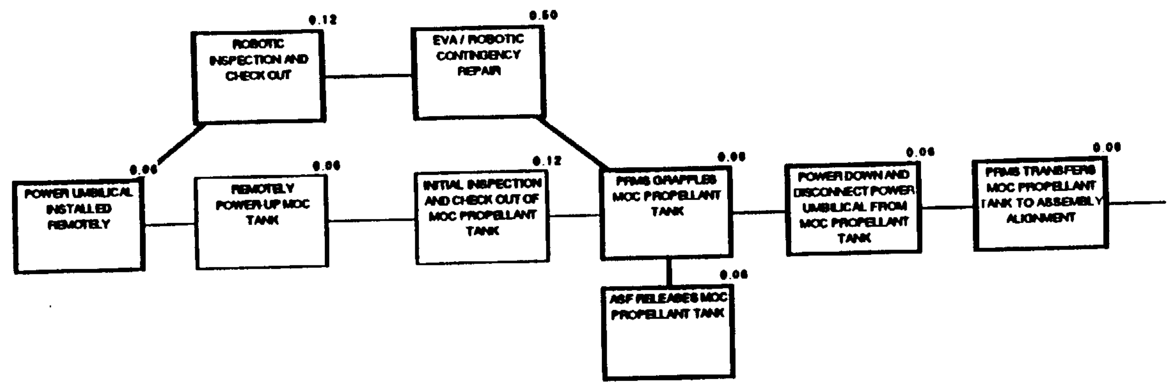

0.06
VERIFY ALIGNMENT TMI PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
PRMS MATES TMI PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.18
AUTOMATIC STRUCTURAL FASTENING OF TMI PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
PRMS RELEASES TMI PROPELLANT TANK
0.12
REMOTE INSPECTION AND CHECK OUT OF MECHANICAL FASTENING
0.06
CONNECT POWER UMBILICAL FROM NTR STRUCTURE TO TMI PROPELLANT TANK
0.50
EVA / ROBOTIC CONTINGENCY ASSEMBLY AND REPAIR
0.12
INSPECTION OF ELECTRICAL/DATA SYSTEMS
0.12
INSPECTION OF FLUID SYSTEMS
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST OF ELECTRICAL/DATA SYSTEMS
0.25
TEST OF FLUID SYSTEMS
0.06
TMI PROPELLANT TANK ASSEMBLY TO STAND BY MODE
0.06
PRMS TO STAND BY MODE

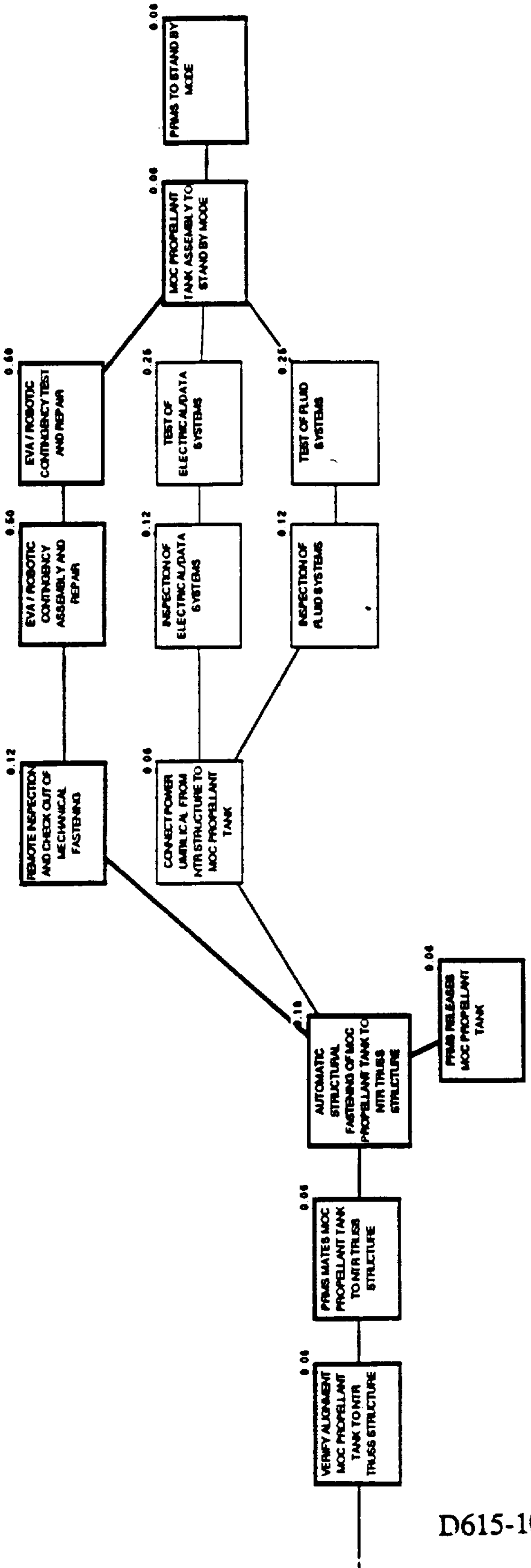
NTR MISSION EIGHT



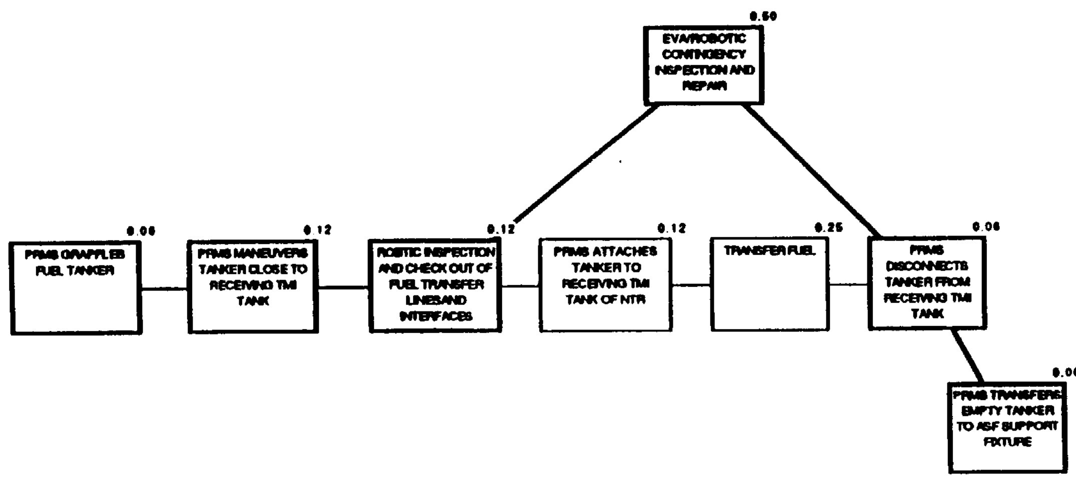

MISSION EIGHT COMPLETE
0.25
CTV RETURNS CREW AND ACRV TO SSF
3
LOW G FUEL TRANSFER FROM TANKER TO TMI TANKS
0.06
PRMS TRANSFERS FUEL TANKER TO ASF STORAGE FIXTURE
0.06
CTV SEPARATES FROM FUEL TANKER
0.06
PRMS GRAPPLES FUEL TANKER
0.25
CTV TRANSFERS FUEL TANKER TO ASSEMBLY PLATFORM
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM

NTR MOC FUEL TANK #2 ASSEMBLY

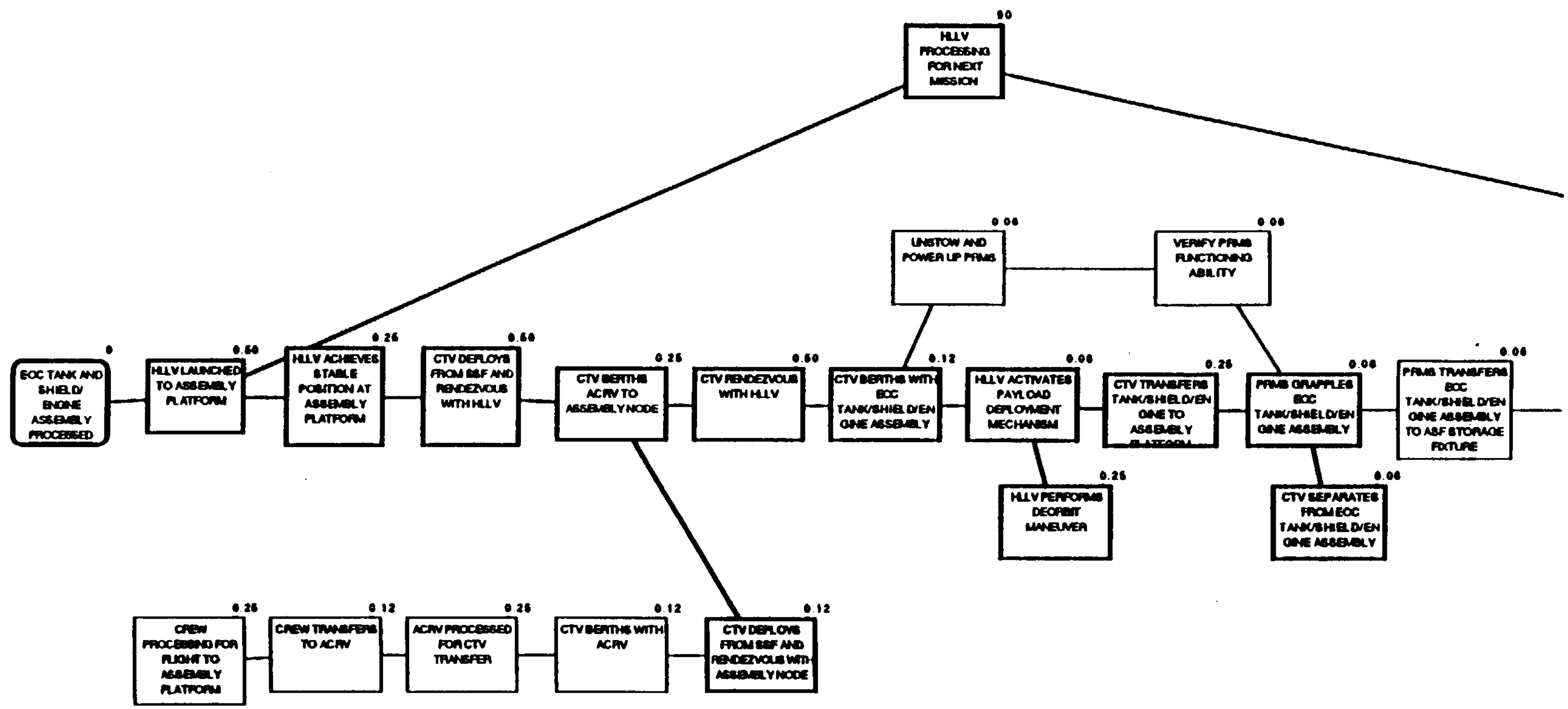


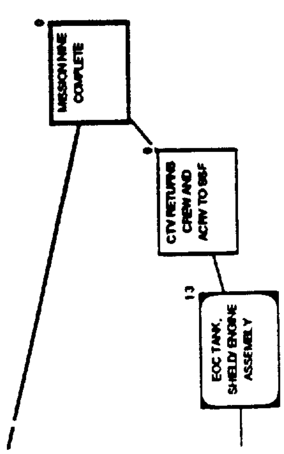

VERIFY ALIGNMENT MOC PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
PRMS MATES MOC PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.06
AUTOMATIC STRUCTURAL FASTENING OF MOC PROPELLANT TANK TO NTR TRUSS STRUCTURE
0.18
PRMS RELEASES MOC PROPELLANT TANK
0.04
REMOTE INSPECTION AND CHECK OUT OF MECHANICAL FASTENING
0.12
CONNECT POWER UMBILICAL FROM NTR STRUCTURE TO MOC PROPELLANT TANK
0.06
EVA / ROBOTIC CONTINGENCY ASSEMBLY AND REPAIR
0.50
INSPECTION OF ELECTRICAL/DATA SYSTEMS
0.12
INSPECTION OF FLUID SYSTEMS
0.12
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.50
TEST OF ELECTRICAL/DATA SYSTEMS
0.25
TEST OF FLUID SYSTEMS
0.25
MOC PROPELLANT TANK ASSEMBLY TO STAND BY MODE
0.06
PRMS TO STAND BY MODE
0.06

NTR OFFLOADED PROP TRANSFER

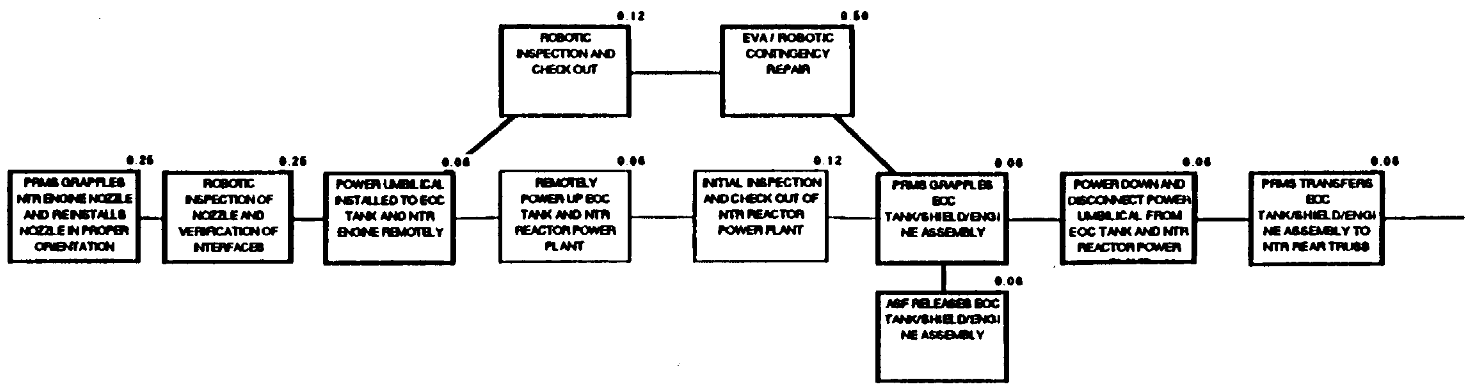


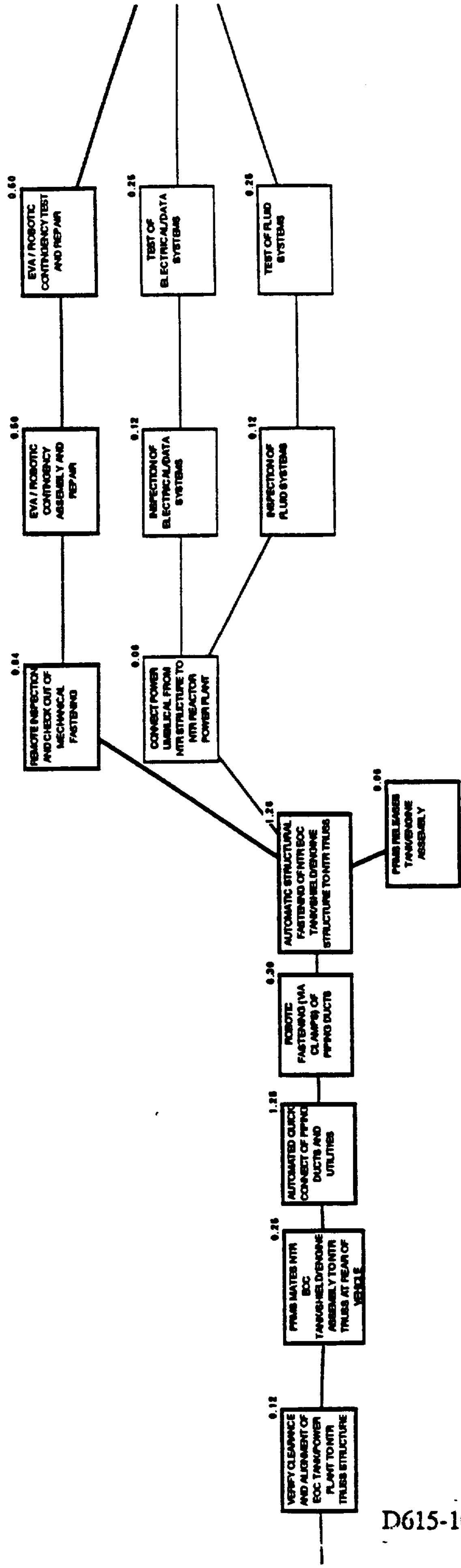
NTR MISSION NINE

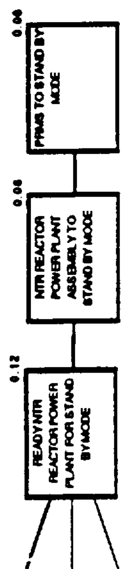



MISSION NINE COMPLETE
CTV RETURNS CREW AND ACRV TO SSF
13
EOC TANK, SHIELD/ ENGINE ASSEMBLY

NTR EOC TANK/SHIELD/ENGINE ASSEMBLY




0.12
VERIFY CLEARANCE AND ALIGNMENT OF EOC TANK/POWER PLANT TO NTR TRUSS STRUCTURE
0.25
PRMS MATES NTR EOC TANK/SHIELD/ENGINE ASSEMBLY TO NTR TRUSS AT REAR OF VEHICLE
1.25
AUTOMATED QUICK CONNECT OF PIPING DUCTS AND UTILITIES
0.30
ROBOTIC FASTENING (VIA CLAMPS) OF PIPING DUCTS
1.25
AUTOMATIC STRUCTURAL FASTENING OF NTR EOC TANK/SHIELD/ENGINE STRUCTURE TO NTR TRUSS
0.08
PRMS RELEASES TANK/ENGINE ASSEMBLY
0.84
REMOTE INSPECTION AND CHECK OUT OF MECHANICAL FASTENING
0.08
CONNECT POWER UMBILICAL FROM NTR STRUCTURE TO NTR REACTOR POWER PLANT
0.50
EVA / ROBOTIC CONTINGENCY ASSEMBLY AND REPAIR
0.12
INSPECTION OF ELECTRICAL/DATA SYSTEMS
0.12
INSPECTION OF FLUID SYSTEMS
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST OF ELECTRICAL/DATA SYSTEMS
0.25
TEST OF FLUID SYSTEMS


0.12
READY NTR REACTOR POWER PLANT FOR STAND BY MODE
0.06
NTR REACTOR POWER PLANT ASSEMBLY TO STAND BY MODE
0.04
PRMS TO STAND BY MODE

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Ground

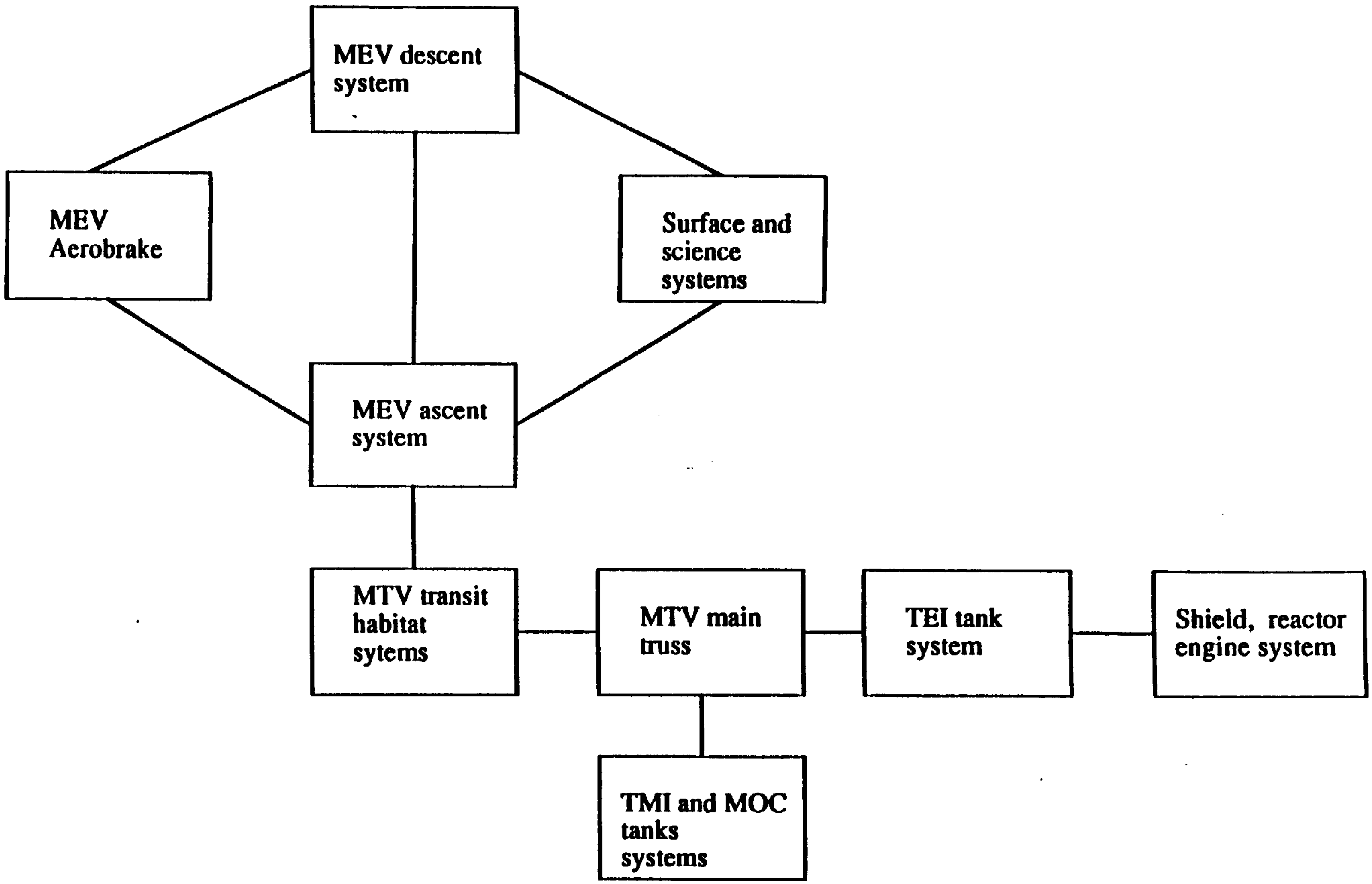
NTR, NEP, and SEP Assembly Flow Summary

- First mission of NTR assembly will require truss to be deployed and secured to dedicated assembly platform to lend additional stability to the platform during vehicle assembly
- The two TMI and two MOC tanks of NTR vehicle are brought up in staggered configuration starting with TMI tank #1 in the fifth HLLV flight. The reason for this is that the off-loaded propellant tankers that come up with the MOC tanks, (flights 6 and 8) will not have to be stored for a prolonged period of time
- The NTR in-line tank (or EOC tank) is integrated with the shield and engine along with associated structures; further the engine nozzle is mounted in reverse to improve packaging efficiency. Portion of reverse-mounted nozzle protrudes into HLLV nosecone space. Engine assembly to the NTR vehicle will first require properly assembling the nozzle to the engine.
- Assembly of NEP Heat Transport and Rejection Systems (Missions 5, 7, and 8) requires nearly the full 90 days between Assembly Flights:
    - Due mainly to number of pieces and connections
    - Increasing number of assembly robots and multi-tasking may reduce this some; however, since this is a serial task, it must be done in steps
    - It is expected that welding pipes, instead of fastening with clamps, may reduce required time (including necessary verification procedures)
- NEP configuration should include robotic access to aft end of vehicle (later configurations include truss for the length of the NEP)
- If ACRV can not accommodate crew assembly operations, some type of control station must exist at assembly site until MTV Hab arrives:
    - ET-based platform devised for Cryo/Aerobrake Vehicle included SSF Node and airlock
    - MTV Hab could come up first (using ground simulators for remainder of interface verification)
- Integrated Aeroshell launch would reduce flights and on-orbit assembly time
- HLLV payload may need to be unloaded in groups rather than individually to prevent violation of HLLV on-orbit stay time

Ground Rules and Assumptions for Ground Processing

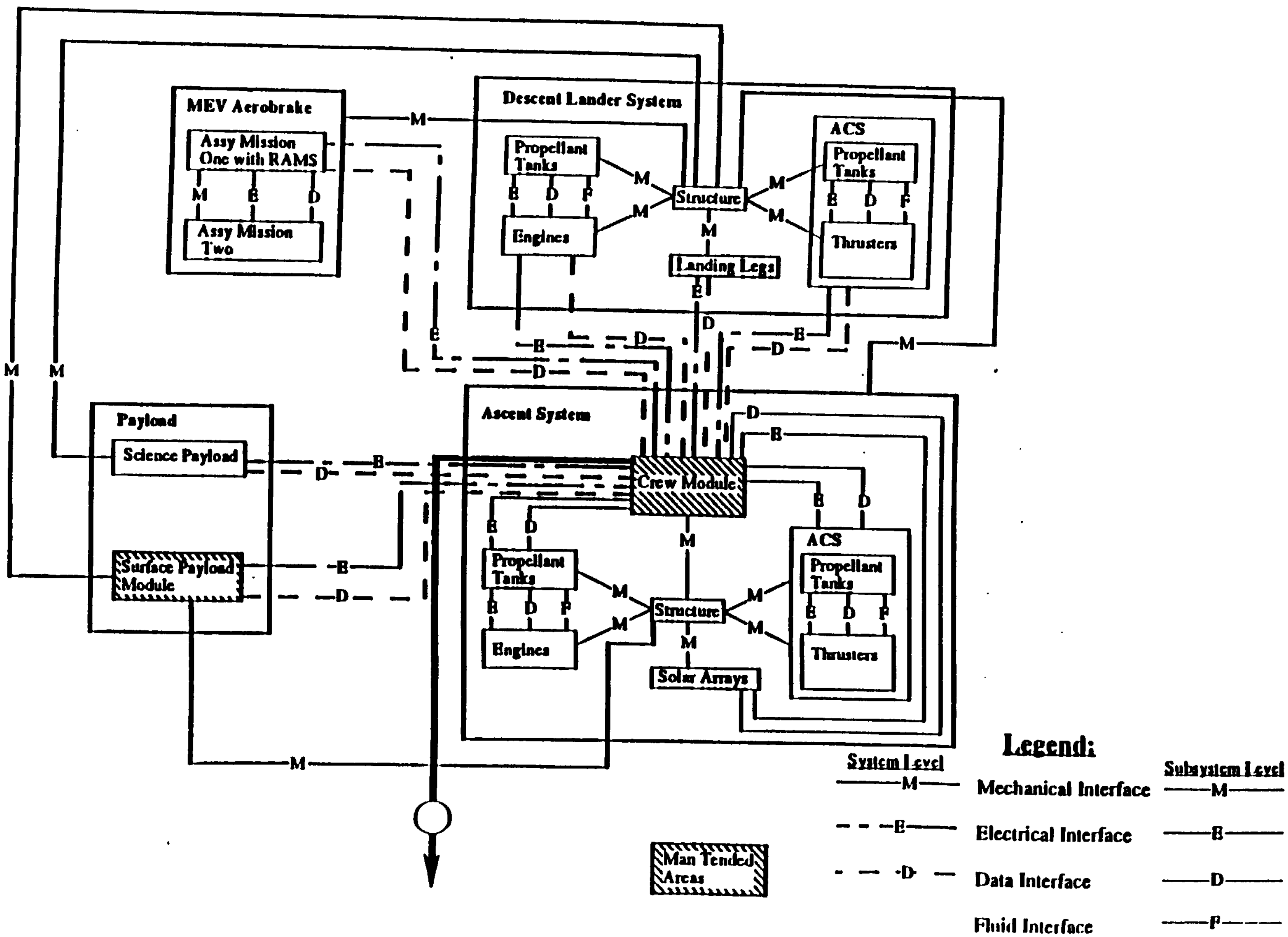
- A system is a group of components and supporting structure that is integrated by a contractor and delivered as a unit to the processing facility (e.g. MEV Aerobrake, MEV descent lander, ascent system, etc.).
- System interfaces are those which transmit data, power, or fluids across the system's boundaries and mechanically secure one system to another.
- Subsystems interfaces are those which are internal to a system.
- Subsystem interfaces are verified by the manufacturer prior to system integration.
- Component interfaces are those which are internal to a subsystem.
- Component interfaces are verified by the manufacturer during subsystem assembly.
- Interfaces verified prior to a system level integration will be accepted with no repetition of tests.
- Flight hardware will be used to verify system interfaces.
- Ground facilities will simulate assembly node operations and limitations.
- Certain non-mechanical interfaces to NTR, NEP, and SEP are simulated to allow desired launch sequencing.

NTR MMV System Interfaces (Top Level)



NTR System Interfaces

MEV System Interfaces



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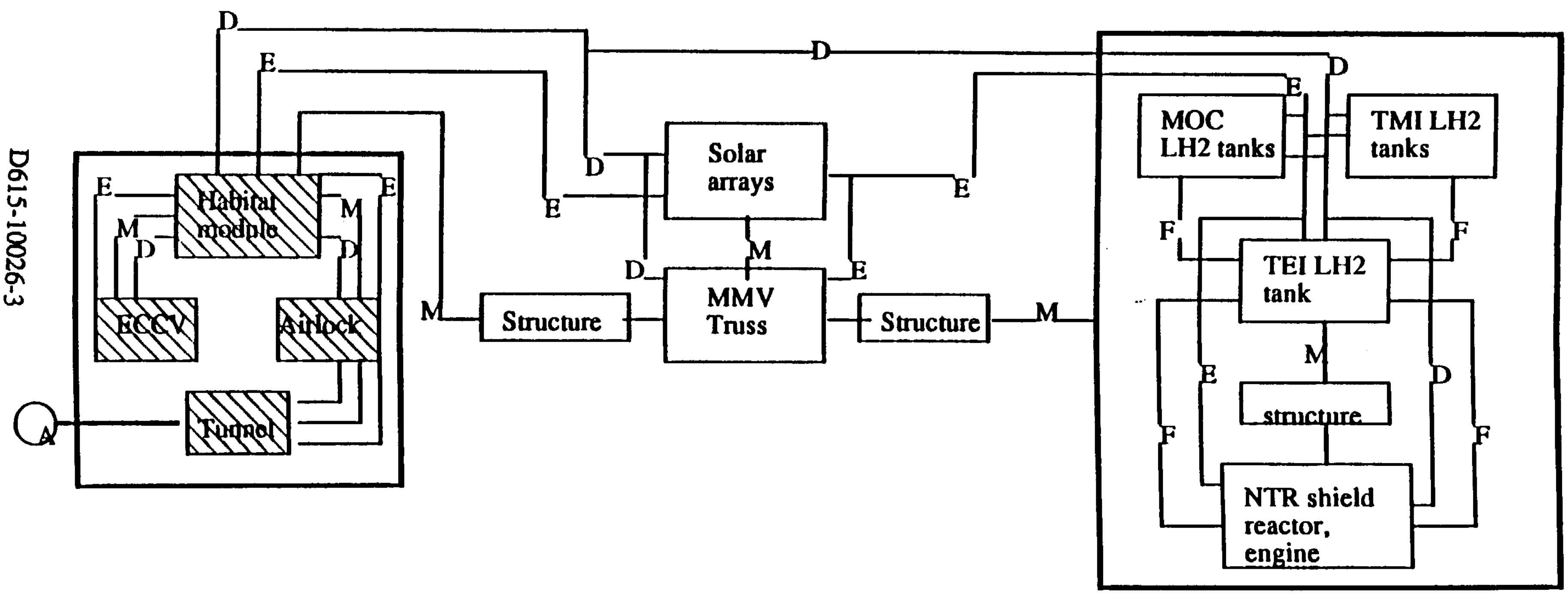
MTV System Interfaces



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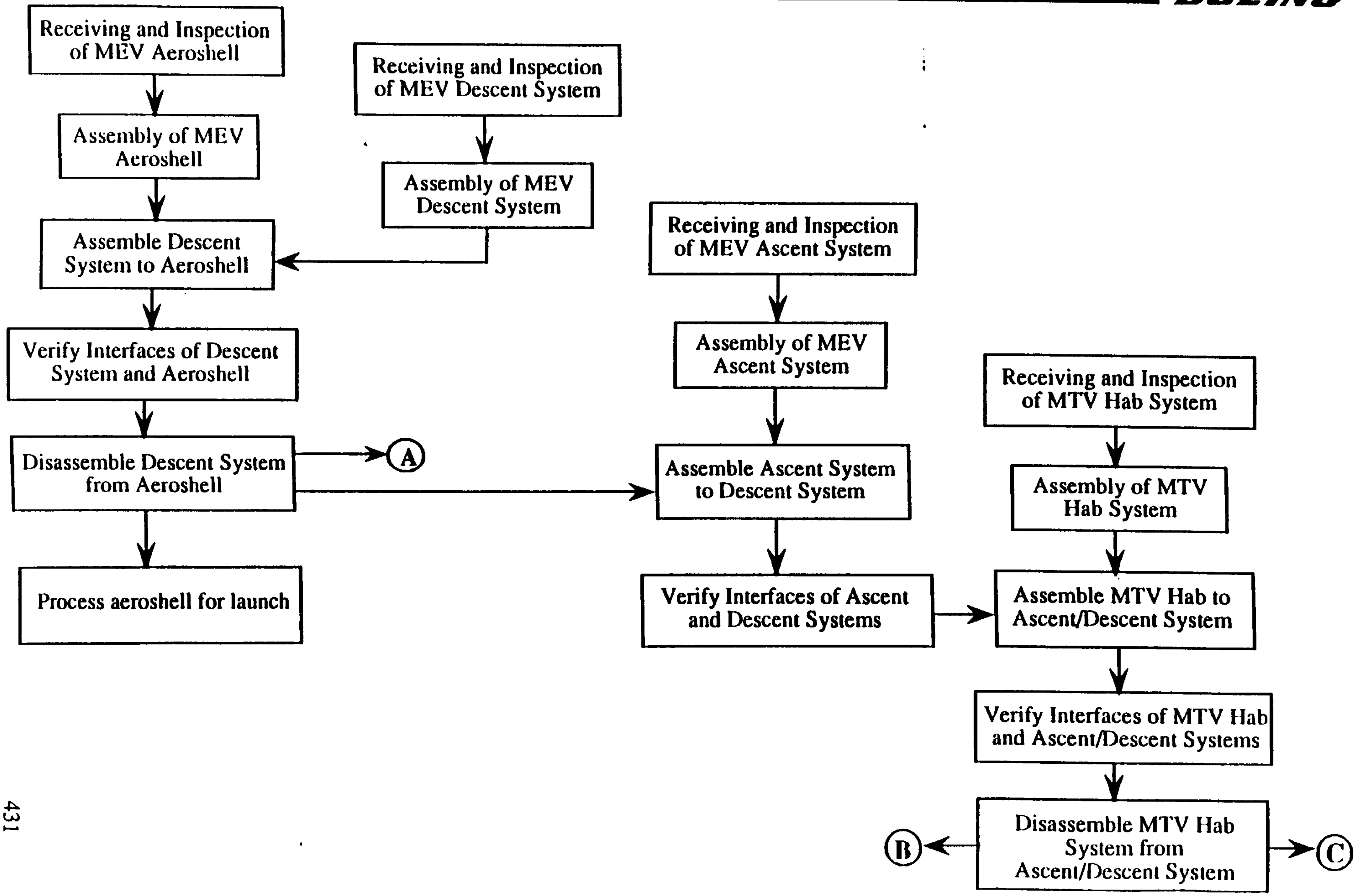
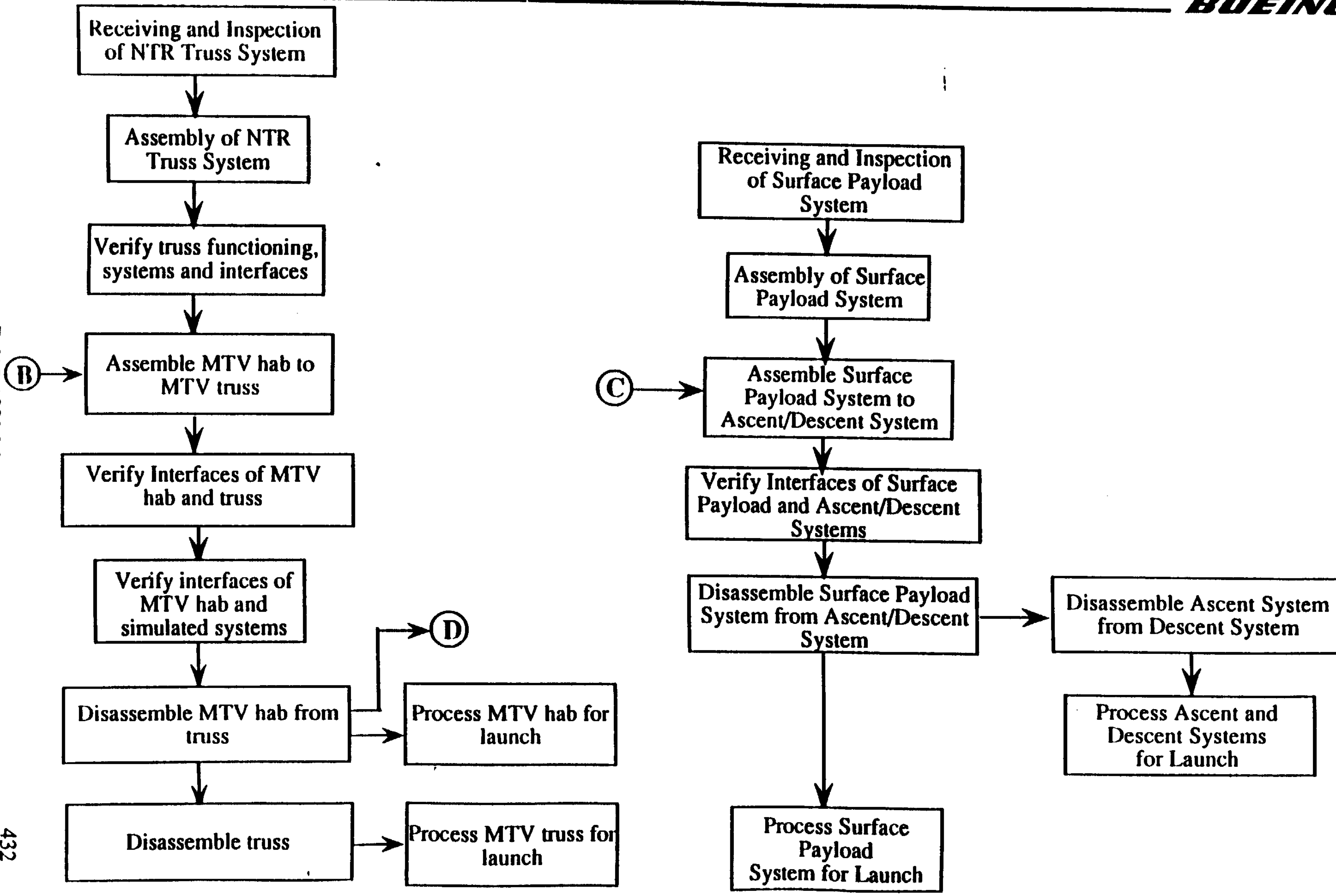
Sequential Interface Verification

- Process of verifying the interfaces of the Mars Mission Vehicles elements without complete assembly.
- Elements are received and inspected at the assembly area.
- Internal test performed and certified by the contractor will not be repeated.
- Elements will be assembled to the level required to verify the interfaces from one element to another.
- Interfaces will be verified by flight hardware when feasible or by match mate devices/prototypes when necessary.
- Elements will be disassembled to payload configurations and processed for launch.

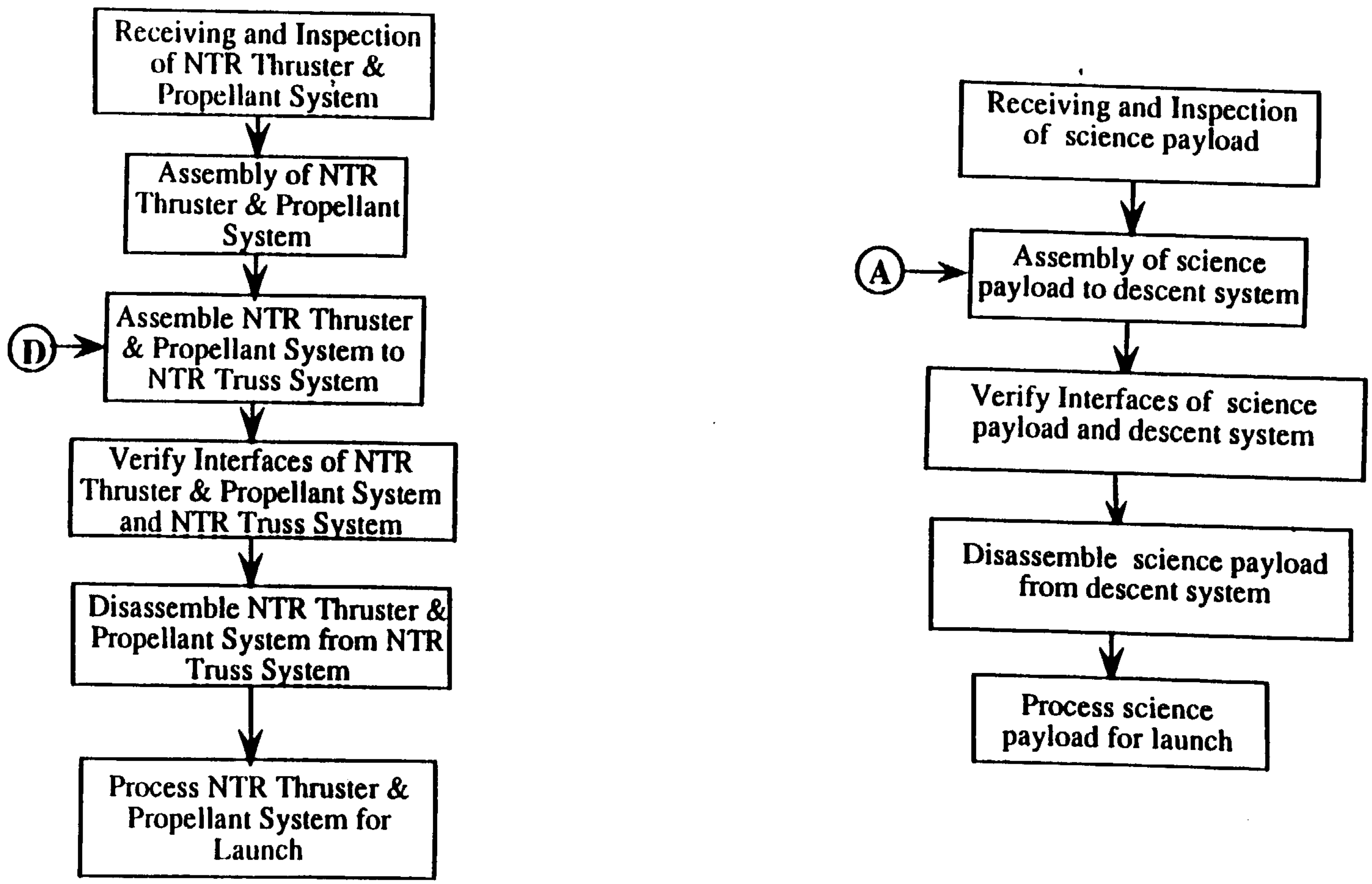
NTR Ground Processing Functional Flow



NTR Ground Processing Functional Flow - continued



NTR Ground Processing Functional Flow - continued



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Special Ground and On-Orbit Processing Facility and Equipment Requirements

Facilities/Equipment	NTR	NEP	SEP
Ground			
• Reactor/engine mating and processing facility	X	X	
• Nuclear fuel loading facility	X	X	
• Contaminated materials storage and disposal facility	X	X	
• Solar array/radiator packing and storage facility	X	X	X
• Alkali metals materials and transferring facility		X	
• Radiation/hazardous materials contamination treatment facility	X	X	
• Robotics to handle radioactive fuels and hazardous chemicals/materials and components	X	X	
• Vehicle truss processing and packaging facility	X	X	X
On-Orbit			
• On-orbit robotic welding and certification equipment	X	X	
• On-orbit alkali metal heating capability		X	
• On-orbit robotic repair/maintenance equipment	X	X	X

Summary (Ground Processing, Manifesting, On-Orbit Assembly)

- Ground processing flows are very interdependent upon the launch vehicle and assembly concept assumed
- Non-hardware interface verification may require simulators to better schedule hardware deliveries
- Assembly flights are mainly volume, not mass, dependent
  - Most flights underutilize relative mass capability
  - A mixed fleet may improve launch packaging efficiency for NEP and SEP
  - Integrated aerobrake launch provides advantage in terms of number of flight and orbital assembly
- Capabilities, requirements of first element launch (FEL) of Mars vehicles drives on-orbit assembly infrastructure
- Two of the NEP assembly stages require nearly the full 90 days allotted between flights
  - Radiators and heat transport system require a large number of operations
  - Changes in assumptions used for number of pieces and method of attachment could easily violate 90 day limit
- Assumed deployable truss for NEP, SEP, and NTR reduces on-orbit times
- Assumed extensive assembly robotics tends to decrease crew time and needed infrastructure

VI. Implementation Plan

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Technology Needs and Advanced Plans

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Technology Issues - NTR

Introduction

Technology issues relating to the NTR vehicle are presented in this section. Some of the charts are also included in the Cryo, NEP, and SEP IP&ED documents. The focus of this section will be to bring out those issues important to the NTR from these charts, and to present a series of technology level requirements necessary for the reference NTR vehicle.The most important technology development needs for NTR are in the areas of nuclear thermal engine development and testing, and low heat leak, minimum mass LH2 tankage.

Technology commonality Issues

The following nine charts lay out the important technology commonality issues between the major propulsion options as well as across the seven major mission architectures identified in this study. The NTR vehicle exhibits commonality to the other vehicles in several important areas. The transfer crew module is substantially the same as the other options. The MEV is identical across all vehicle options, except for the cryogenic propellant management and storage system, which must provide storage for the outbound trip, instead of transferring it from larger tanks prior to landing (at least for O2). The LH2 storage and propellant management system design will be similar to that necessary for the chemical vehicle. The demands placed on the avionics system for the NTR system are similar to those for any high thrust system. Finally, in-space assembly issues should be similar for the NTR and cryo/aerobraked vehicle, with the exception of the related nuclear issues associated with the NTR.

The seven identified Lunar/Mars mission architectures verses the required component technologies, enabling and enhancing, are shown on the next set of charts and facing page text. Many of these component technology issues are common across the listed architectures. These issues are for the entire integrated architectures, and do not necessarily refer specifically to the NTR vehicle. The areas of multi-MW nuclear thermal energy production, and high temperature fuel element materials are the primary areas of technology development concern for the NTR option. Commonality to the initial cryogenic vehicles could enhance the NTR as a growth option, while the near-term nature of the related technologies would qualify it as an alternative to a cryogenic/aerobraked mission.

Technology Development Concerns

As noted before, many of the identified critical and high leverage technology development issues are common across all four major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H2, and possibly O2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique NTR technology issues include high temperature fuel element materials, high power reactor advanced development, and reactor shielding. Enhancing technologies include cryogenic refrigeration (lander tanks), O2-H2 RCS, advanced in-space assembly techniques, and advanced structural materials development.

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NTR Vehicle Technology Requirements

Technology performance levels required for the NTR reference vehicle are outlined in the next six charts. These are not intended to be the levels needed for a minimum NTR vehicle, but serve mainly to document the levels required to accomplish the identified reference mission profile with the vehicle model as configured. Changes to these specifications would not necessarily affect the feasibility of a NTR mission, but would change the reference vehicle configuration. The list also includes operational requirements which could drive technology development or advanced development. An example of this could be the required engine gimble angle which would drive reactor design.

NTR Technology Development Schedule

The final chart in this section is a proposed technology development schedule for the nuclear electric propulsion option. The schedule shows that, given a FY '91 start, the SEP vehicle could be ready for a Mars mission in the 2009 timeframe. A full scale decision point is also highlighted during year 7. This is the point where a commitment should be made for full scale funding and development of the program.

Technology Commonality and Differences

System/Subsystem	Reference (Cryo-A/B)	NTR Vehicle	NEP Vehicle	SEP Vehicle
Crew Systems/Habitats Life Support, rad. prot., hab. struct., & airlock/EVA	Long duration life support system derived from SSF proven system. LTV crew module evolves to MTV; common LEV/MEV habitat system. Mars surface habitat derived from proven Lunar design. Mars surface TCS requires additional technology advances to deal with unique heat rejection problems. All extended missions (>2-3 d) require solar flare radiation protection. Hab systems common across mission architecture. Shorter mission LSS sized for free return abort contingency. Minimum mass airlock could be shuttle-evolved.			
Power System & Thermal Control	Deployed solar array system; low power (~50-75 kW). Low temp heat rejection (~400°K)	Common to reference vehicle system	Nuc. /Rankine or Brayton cycle energy conv. sys. Very high power level (up to 200 MW). High temp heat rejection (~1000°K-main cycle).	Solar-electric energy conversion. High power (~10 MW or greater) level. Moderate temperature radiators (400 - 650 K).
Propellant Management & Storage	Long term storage of H_2 & O_2 for Earth & Mars orbit, and deep space environ. necessary with minimal boiloff. Low-g fluid gaging, acquisition, and transfer highly enhancing or enabling for all missions. NTR requires common techniques for LH_2 fuel.		Argon propellant management system can be similar to LOX storage system, but without the safety constraints associated with an oxidizer.	
Propulsion System	Advanced cryogenic space engines with >475 sec Isp, and ~30 klb to ~200 klb thrust.	NERVA derived /advanced NTR system with higher Isp (up to 1050 sec vs. 850 sec.)	Rankine or Brayton cycle conversion system driving cluster of Ion thrusters for NEP. Same thrusters for SEP. Number of thrusters depends on available thruster size and required redundency.	
Aerobraking Lunar	Low L/D - AFE derived for Earth capture.	Not needed for Lunar NTR (propulsive capture@ Earth)	Not needed for NEP or SEP.	
Mars	Higher L/D necessary - structure and TPS technology base.	Only low energy lander aerobrake needed, since entire vehicle, including MEV is propulsively captured at Mars. Can be common with earlier cryo A/B vehicle, unless crossrange constraints require higher L/D design.		
Avionics	Avionics system hardware may be common for Lunar or Mars (or L/M growth)		Avionics system required for low & continuous thrust vehicles are lower than for Cryo A/B or NTR vehicle.	
Assembly & Checkout	Common assembly facility & equip. for most mission vehicles. Assembly time in LEO, and thus M/D protection level is varied. Mars vehicle requires launch & assembly of large (~ 30 m vs. 20 m for Lunar) aeroshell. Nuclear vehicles (NTR & NEP) may face political constraints on launch & assembly of vehicle. Assembly & operation may be necessary from nuclear safe orbit.			Severe LEO debris environ. damaging to solar arrays. Spare set of arrays may be necessary. MEV A/B launch & assembly needed.

Required Technologies vs. Alternative Mission Architecture

A set of required technologies for the seven identified alternative mission architectures outlined in the evolotionary concepts section is presented. The purpose of this matrix is to provide a preliminary comparison of technology development needs for the alternative architectures. The matrix also serves to better define the architectures. From this top level matrix, a more detailed set of technology requirements can be derived. A set of accommodating technologies can be compiled for needs areas where options exist. Finally, the technology areas can be prioritized as enabling and enhancing, and a return on investment performed for identified high leverage technologies. This portion of the matrix includes most of the cryogenic management issues. Enabling technologies are represented by the filled circle, and enhancing technologies by the open circle. Extensive low - g cryogenic propellant launch, acquisition and transfer refers to the Mars conjunction case, and the mass driver option, where propellant will be used for the transfer vehicles, which will be parked in a low - g environment (Lunar or Mars orbit, or libration staging point). The Mars cycler orbit case includes a question mark for the long term cryogenic storage system, because the necessary thrust levels and type of propulsion system are undetermined at this time.

Required Technologies vs. Alternative Mission Architecture

	Low boiloff cryogenic propellant storage system (1-3 yr)	Low boiloff cryogenic propellant storage system (15 - 60 d)	Low - g fluid acquisition and transfer	Extensive low - g cryogenic propellant launch, acquisition, and transfer	Cryogenic tank integrity monitor	Cryo fluid rematible umbilical	Lunar LOX production, liquification, and transfer technology	Mars O2 production, liquification, and transfer technology
Mars NEP Alternative Architecture	○	●	●		●	●	●	●
Lunar/Mars NTR Alternative Architecture	●	●	●		●	●	●	
Mars SEP Alternative Architecture	○	●	●		●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●	●	
Mars Cycler Orbits Alternative Architecture	?	●	●		●	●	●	
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●	●	● + H2
Lunar / Mars NEP Alternative Architecture	○	●	●		●	●	●	●

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix section represents the major aerobraking concerns. The aerobraking energy columns for Mars and Earth capture digresses from the format in order to illustrate the energy levels, and therefore, the level of technology development needed for the various architectures. Aeroheating predictions, reusable aerobrake TPS, advanced GN&C, and TT&C follow along with the high and medium energy missions. Again, a question mark is shown for the Mars cycler orbit case. Reusable TPS for Earth return cannot be determined as a technology development concern until the aeroheating load at Mars can be determined for the cycler orbits. Further mission design efforts must be carried out before an estimate on this can be made.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Earth return aerobrake energy	Mars capture aerobrake energy	Mars lander aerobrake	High performance aerobrake structure	Aerobrake assembly and test	Aeroheating prediction (Earth and/or Mars)	Reusable aerobrake TPS for Earth return	GN & C to protect TPS	Advanced high accuracy and rate TT & C	In space AR&D / assembly
Mars NEP Alternative Architecture	Low	Low	●	●	●					●
Lunar/Mars NTR Alternative Architecture			●	●	●					●
Mars SEP Alternative Architecture	Low	Low	●	●	●					●
L2 Node / Mass Driver Alternative Architecture	High	High	●	●	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	High	High	●	●	●	●	?	●	●	●
Mars Conjunction/Direct Alternative Architecture	Medium	Medium	●	●	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	Low	Low	●	●	●					●

● - Enabling

O - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix area represents the major propulsion issues, with the exception of the radiation protection system, for the baseline and alternative mission architectures. The system to inert and can waste for radiation shielding can be enhancing, while a GCR and ALSPE shelter is enabling for all mission architectures. Again, due to the undefined Mars cycler orbit trajectories, it is questionable as to the need for a large cryogenic space engine. A H_2 - O_2 ACS/RCS system is noted as enabling for each option, as it will be for any option over a baseline storable system. A Lunar orbital momentum storage and transfer device such as a bolo can be enhancing for all missions, after an initial launch and assembly penalty for the massive (~ 1000 Mt) device.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Large (150 - 200 klb) cryogenic advanced space engine	Small (15 - 30 klb) cryogenic advanced space engine	H2 - O2 ACS/RCS	Multi - MW space based nuclear electric power	Multi - MW space based nuclear thermal power	Surface nuclear electric power	Multi MW solar power system (arrays and handling equip.)	Radiation protection (system to inert & can waste)	Mass driver / rail gun technology	Lunar orbital momentum transfer device (Bolo)
Mars NEP Alternative Architecture		●	○	●		●		●		○
Lunar/Mars NTR Alternative Architecture		●	○		●	●		●		○
Mars SEP Alternative Architecture		●	○				●	●		○
L2 Node / Mass Driver Alternative Architecture		●	○			●		●	●	○
Mars Cycler Orbits Alternative Architecture	?	●	○			●	●	●		○
Mars Conjunction/Direct Alternative Architecture	●	●	○			●		●		○
Lunar / Mars NEP Alternative Architecture		●	○	●		●		●		○

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

The final section of the matrix is not as illustrative as the others, in that all of the listed technologies are enabling, with the exception of a closed ecological life support system, which is significantly enhancing for all identified mission architectures.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Autonomous health monitoring and check - out	High data rate comm. or high performance compression	DMS/system diagnostics. Art. intell/neural nets/high processing rate GN&C	Long duration refurbishable crew habitat	Long duration BCLSS	CBLSS
Mars NEP Alternative Architecture	●	●	●	●	●	●
Lunar/Mars NTR Alternative Architecture	●	●	●	●	●	●
Mars SEP Alternative Architecture	●	●	●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	●	●	●	●	●	●
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	●	●	●	●	●	●

Mars NTR Vehicle Technology Requirements

I. TMIS / MTV

A. Cryogenic storage system

1. Thermal protection system: TMIS - MLI over foam. (1" foam; ~ 1" MLI) MTV - MLI; MLI over foam; ~100 layers MLI over ~1/2 - 1" foam.
2. Tanks launched wet.
3. Thermodynamic vent coupled to a single vapor cooled shield.
4. Topoff before Earth departure.
5. ~ 6 months in LEO before use.
6. Negligible boiloff loss after topoff.

B. Propulsion

1. Isp = 925 - 1050 s.
2. Thrust = 10 - 250 klb/engine.
3. Solid core reactor NTR engine.
4. Burn lifetime up to 10 hr (varies w/Isp).
5. No throttling requirements.
6. Gimbal angle (nominal) = 10°
7. Space exposure life = 10 yr.
8. Chamber pressure: 450 - 1000 psia (high press.), <10 psia (low press.)
9. Capable of mixed phase flow starts.
10. In-space changeout capability.
11. Off vehicle preflight checks.
12. No retraction / extension required.

Mars NTR Vehicle Technology Requirements (cont.)

C. Structure
1. Material - metal matrix composites, advanced alloys, and organic matrix composites.
2. Meteor/debris protection provided for tanks and plumbing.

D. Avionics
Piggybacked on MTV.

E. Power
1. Level : < 1 kW (electrical system only)
2. System: Auxiliary power units on engine pod; piggybacked on MTV for back-up.

F. Assembly
1. Off station assembly.
2. Degree of assembly: Separate tanks connected to primary structure in LEO to form propulsion stage.

G. Habitat
1. ECLSS: Space Station Freedom derived system with similar degree of closure; potable H_2O from cabin condensate; CO_2 reduction/regeneration; Hygiene H_2O from urine processing. CELSS to be evaluated.

Mars NTR Vehicle Technology Requirements (cont.)

2. Structure
   a. 2219 - T8 aluminum pressure vessel.
   b. Pressurized to 20 psig on launch for structural integrity.
   c. Insulation & M/D shield external to pressure shell.
   d. No penetrations in end domes.
   e. Radiation storm shelter provided, and configured to utilize equipment & supplies as partial shielding.
   f. External space radiator integral with M/D shield.

3. Cabin repressurizations: 2+ (outbound emergency could use propellant for repress.)
4. Spares: 15% of active equipment - component level.
5. Redundancy: Two complete and separate systems for life critical systems + spares. Component changeout capability.
6. Residence time = 535 days.
7. Science: Transit science as allowed by individual mission.
8. EVA capability: EVA suits provided for all crew; EVA waste fluid recovery for ECLSS.

H. <u>ECCV</u>

1. Apollo size & style.
2. Open ECLSS (LiOH, no H_2O recovery).
3. Residence time: 2 - 3 days.
4. Propulsion: RCS only.

Mars NTR Vehicle Technology Requirements (cont.)

II. MEV

A. Cryogenic storage system

1. Thermal protection system: 100 layers of MLI for H_2 and O_2 tanks (2").
2. Tanks: double wall tanks with vacuum annulus;
   low thermal conductivity support system for inner tank.
3. Thermodynamic vent: Simple design for gravity field.
4. Tanks launched dry and filled prior to descent, from MTV tanks, or refrigerated. (no boiloff prior to descent)
5. Stay time from 30 - 600 days on Mars surface.
6. Boiloff level < 20% for surface stay.

B. Propulsion

1. Isp = 460 sec.
2. Thrust = 30 klb / engine.
3. Nozzle area ratio = 200.
4. Throttleability = 15:1.

Mars NTR Vehicle Technology Requirements (cont.)

B. Propulsion (cont.)
- 6. Gimbal angle (nominal) = 10°.
- 7. No restart capability necessary for nominal case.
- 8. Space storage time between burns : NA.
- 9. Engine out capability (crossfeed propellant lines).
- 10. Expander cycle.
- 11. In-space changeout capability.
- 12. Off vehicle preflight checks.
- 13. Retraction / extension capability.

C. Structure
1. Vehicle
   - a. metal matrix composites / advanced alloys / organic matrix composites.
   - b. Micrometeoroid protection for tanks and plumbing.

2. Aerobrake
   - a. L/D = 0.5 to 1.0
   - b. Crossrange: 1000 km.
   - c. Vhp = 7.07 km/sec^2.
   - d. Maximum g loading: 6.
   - e. Maximum temp: TBD (estimated 3100° F).
   - f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
   - g. TPS material: Advanced reradiative tiles.
   - h. Relative wind angle (reference) = 20°.

Mars NTR Vehicle Technology Requirements (cont.)

D. Avionics
   1. Error without beacon = 1 km.
   2. Touchdown error = 1 m/s.
   3. Obstacle avoidance capability.

E. Power
   1. Level: ~ 2.5 kW.
   2. System: fuel cells (regenerable).
   3. Back-up system: abort to orbit.

F. Assembly
   1. Off station assembly.
   2. Assembly level (complexity): TBD

G. Habitat
   1. ECLSS: open system; stored potable H_2O; LiOH CO_2 adsorption.
   2. Structure
      a. Aluminum (2219 - T8) pressure vessel.
      b. Overpressurized on launch for structural integrity.
      c. Insulation and micrometeoroid protection external to pressure vessel.
      d. No penetrations in end domes.
      e. No radiation shelter provided in MEV.
      f. External space radiator integral with micrometeoroid shield.
   3. Repressurizations: 2.
   4. Spares: 15% of active equipment mass; component level.
   5. Redundancy: EVA suits as backup to cabin repressurization.; no system level ECLSS redundancy required due to low complexity open system.
   6. Residence time: ~3 days (surface systems support surface stay).
   7. Science: none.
   8. EVA capability: provided for all crew; transferred from MTV.

Critical Lunar/Mars Reference Technology Development Concerns

A preliminary set of critical technology development concerns was constructed for the Lunar/Mars reference missions. Its purpose is to show a top level representation of the areas which could prove enabling for the reference Lunar and/or Mars missions, without further concentrated research and development, flight testing, and/or precursor missions. Aerobraking may prove enabling for most Lunar and Mars missions, and significantly enhancing for the rest, primarily due to reduced demands on limited Earth to orbit launch capability and lower launch costs. Aeroheating prediction codes cannot be validated without further experimental data (flight or ground simulation data). The degree of development needed for aerobrake TPS materials will be determined by these predictions. Low gravity human factors, to be evaluated on SSF, may affect vehicle design significantly. Foe example, vehicle designs must accommodate artificial - gravity until a need level can be determined from space station based research. Finally, precise mission design, incorperating advanced tracking, telemetry, and GN&C must be verified to accommodate aerobraking and automated rendezvous & docking requirements.

Critical Lunar/Mars Reference Technology Development Concerns

Technology	Comments
High Energy Aerobraking - Thermal protection - High performance structure - Theoretical code validation - Deep space tracking, telemetry, and communication	- Heating rates greater than seen by AFE for Mars cap. and Mars/Earth return. - High temp reradiative or lightweight ablative materials needed - Precursor missions needed for existing aeroheating/GN&C codes - 17 minute Mars/Earth comm delay will dictate internal GN&C system.
Advanced Space Engine Development - Large engine (150 - 200 klb thrust) - Small engine (15 - 30 klb thrust; throttleable)	- High thrust, high Isp cryogenic engine for TMI stage. - Low thrust, high Isp, throttleable engine for Lunar/Mars descent and ascent.
Low - g Human Factors	- Vehicle designs should accommodate artificial-g configuration until SSF based life sciences research can be carried out.
Autonomous System Health Monitoring	- Reliable autonomous systems with self monitoring, diagnostic, and correcting capability.
Long Term Cryogenic Storage and Management	- Advances in long term low - g cryo fluid storage and management required for Lunar/Mars initiatives. - low - g propellant acquisition and gaging enabling for all cryo missions.
Long Duration, High Degree of Closure ECLSS	- Reliable SSF validated ECLSS equipment critical for early long term missions.
Efficient Radiation Storm Shelter Material & Configuration	- Improved solar flare prediction/detection, with storm shelter designs incorporating effective lightweight materials - Reliable radiation dosimetry techniques also important
In - Space Assembly; AR & D	- Large aerobraked vehicles will require large degree of in - space assembly. - AR&D critical for both Lunar/Mars orbital operations.

Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

A preliminary set of high leverage technologies was assembled for the Lunar/Mars reference missions. These technologies are enhancing for most, and in some cases, all identified mission architectures. Aerobraking will be significantly enhancing for all Lunar and Mars missions where it is not identified as enabling. Other aerobraking issues which could prove enhancing are lightweight reradiative or ablative TPS material, and ECCV vs. aerocapture of MTV at Earth. Low - g propellant handling and low boiloff cryogenic storage are also very enhancing for any missions where it is not enabling. Advanced propulsion options such as NTR, GCR, SEP, and NEP may prove to be high leverage technology options to baseline cryogenic propulsion systems. Finally, developments in advanced materials can be significantly enhancing in a variety of areas.

Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

Technology	Comments
Aerobraking - Mars Capture (vs. propulsive cap.)	- Aerocapture at Mars can reduce IMLEO >50% over propulsive capture
Aerobraking - Earth Capture (vs. ECCV)	- ECCV reduces IMLEO and thermal protection system (TPS) requirements. - Reusable MTV can reduce life cycle cost.
Aeroshell TPS (reradiative vs. ablative)	- Reusable aeroshell requires rerad. TPS at Mars (or thick lightweight ablator), and ablative at Earth. - Further materials and processes advances or low energy mission may allow Earth/Mars reradiative TPS.
Advanced Long Term Cryogenic Storage Technology	- Cryogenic boiloff reduction technologies such as advanced MLI design and application, VCS, para to ortho H2 conv., and thermal disconnect struts, can reduce IMLEO significantly with low R & D effort - Longer missions offer greater IMLEO savings potential
Low - g Propellant Transfer	- Low - g propellant transfer technology enhancing for all Lunar/Mars mission arch., and enabling for some Lunar missions.
Efficient Cryogenic Refrigeration System	- Cryogenic refrig system can reduce vehicle mass and enhance system reliability at the expense of an increased power level.
O2 - H2 ACS / RCS	- O2 - H2 ACS/RCS (Isp = 400 s) reduces system mass over lower Isp storables
High Isp Advanced Space Engine	- High Isp advanced space engine (Isp = 485 s) enhances all mission phases for all mission arch.
NTR Propulsion System	- NTR propulsion system for the TMI, Lunar transfer, and Mars transfer stages
Advanced In - Space Assembly Techniques	- Launch vehicle capability drives on - orbit assembly level. - Degree of on - orbit assembly capability affects vehicle configuration, ground assembly/processing, and launch manifesting.
Advanced Materials Development	- Advanced materials such as metal and organic matrix composites reduce system inert mass, strength, and/or manufacturing costs. - Some advanced M&P may prove enabling for some mission arch. (ex:Mars/ Earth capture aerobrake)

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Schedules

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Technology Development Concerns and Schedules - Nuclear Thermal Propulsion (NTP)

Critical technology development issues relating to the reference NTP vehicle are presented in this section. Where applicable, the same charts are also included in the CAB, CAP, NEP, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference NTP vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Nuclear Thermal Propulsion Technology Development

The most important area of technology and advanced development for this vehicle option is the development of an integrated nuclear thermal propulsion system. A preliminary schedule for the development of a NTP system for a Mars vehicle is presented. The schedule highlights both the point where a full scale development decision can be made (year 5), and when the first flight article will be available to the vehicle program (year 14). The largest single technology development challenge for the program will probably be test facility design and development. The NERVA program nuclear tests were carried out in a testbed facility open to the atmosphere. Any future test facility must be closed in order to contain the fission products contained in the exhaust gasses. A scrubbing system must be included to remove the fission products from the exhaust gas before it can be released into the atmosphere. This facility may prove to be very costly to build and operate. Nuclear thermal propulsion should offer a shorter development time than the other advanced propulsion options (NEP, SEP), with significantly better performance than the chemical options. The major reactor technology issues are high temperature fuels, efficient frit design, fuel burnup, and nuclear safety issues.

Cryogenic Fluid Management

The large amounts of Hydrogen required for NTP Mars missions increases the importance of technologies development relating to cryogenic fluid management and storage. A preliminary technology schedule is presented for cryogenic fluid system development for Mars mission applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected component fluid management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem development (selected components) and verification/validation tests. Many of the technology issues will be answered during the technology/advanced development work to be carried out for a Lunar program. The major technology obstacles to be overcome by an NTP storage system are in the areas of tankage mass minimization and large scale (relative to Lunar) storage systems development, integration, and orbital/flight operations (fluid transfer, acquisition, etc.).

Vehicle Avionics and Software

Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is produced, however. A technology development schedule for advanced communications is presented.

Life Support

A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.

Aerobraking (low energy)

Low energy aerobraking will offer mission benefits in the areas of decreased demands on the descent propulsion system, and improved crossrange capability. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems (although not as high as for high energy aerobraking), avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiating materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space assembly and ops problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule. It should be noted that this schedule was built with high energy aerobraking in mind, and will possibly be compressed to some degree if only low energy aerobraking is developed.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MEV aerobrakes. As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence.

Summary

As noted before, many of the identified critical and high leverage technology development issues are common across all of the major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique NTP technology issues center around nuclear reactor and engine systems development. Common enhancing technologies include cryogenic refrigeration (lander tanks), O2-H2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

Advanced Propulsion Technology Development Schedule - NTP

A proposed development schedule is presented for a nuclear thermal propulsion (NTP) system. The schedule is for a representative advanced NERVA concept (Isp = 925 s). Liquid or vapor core reactor systems will require significantly longer. The schedule includes both technology and advanced development tasks necessary to produce an initial flight article for the flight program in year 14. The years are listed sequentially, so the schedule can be inserted into the appropriate initial year of a given program schedule. This schedule was integrated into the overall program schedule developed for the Lunar/Mars full science scenario (median scale). Timelines for the development of requirements, system designs, test facilities, components, and integrated systems are included in the schedule. Required system level tests are also included in the schedule, which continue past the first flight article delivery at the conclusion of the DDT&E effort. The later tests will be oriented towards system reliability and performance improvements for later production flight units.

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Preliminary SEI Technology Development Schedules

ADVANCED CIVIL SPACE SYSTEMS

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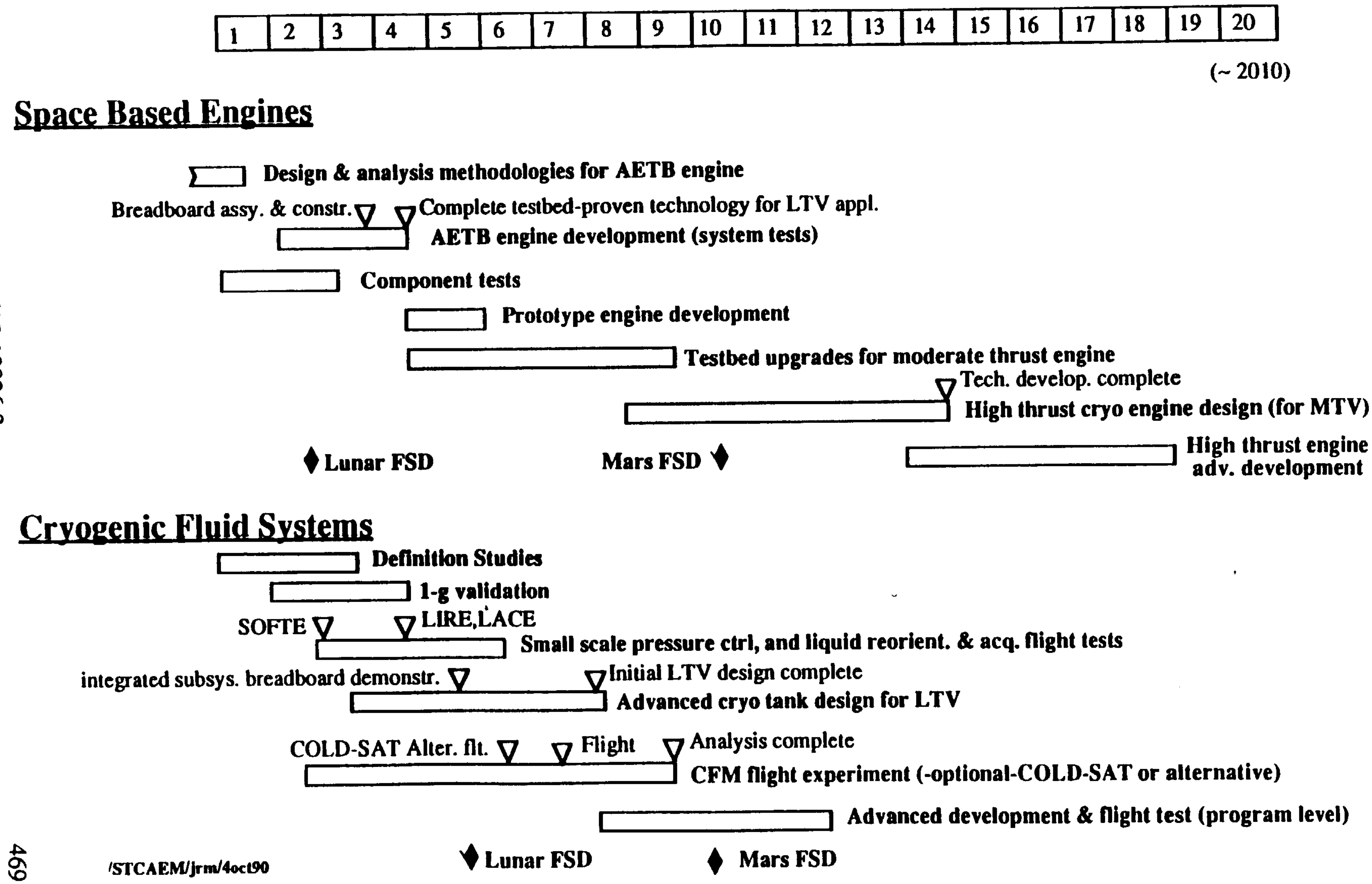

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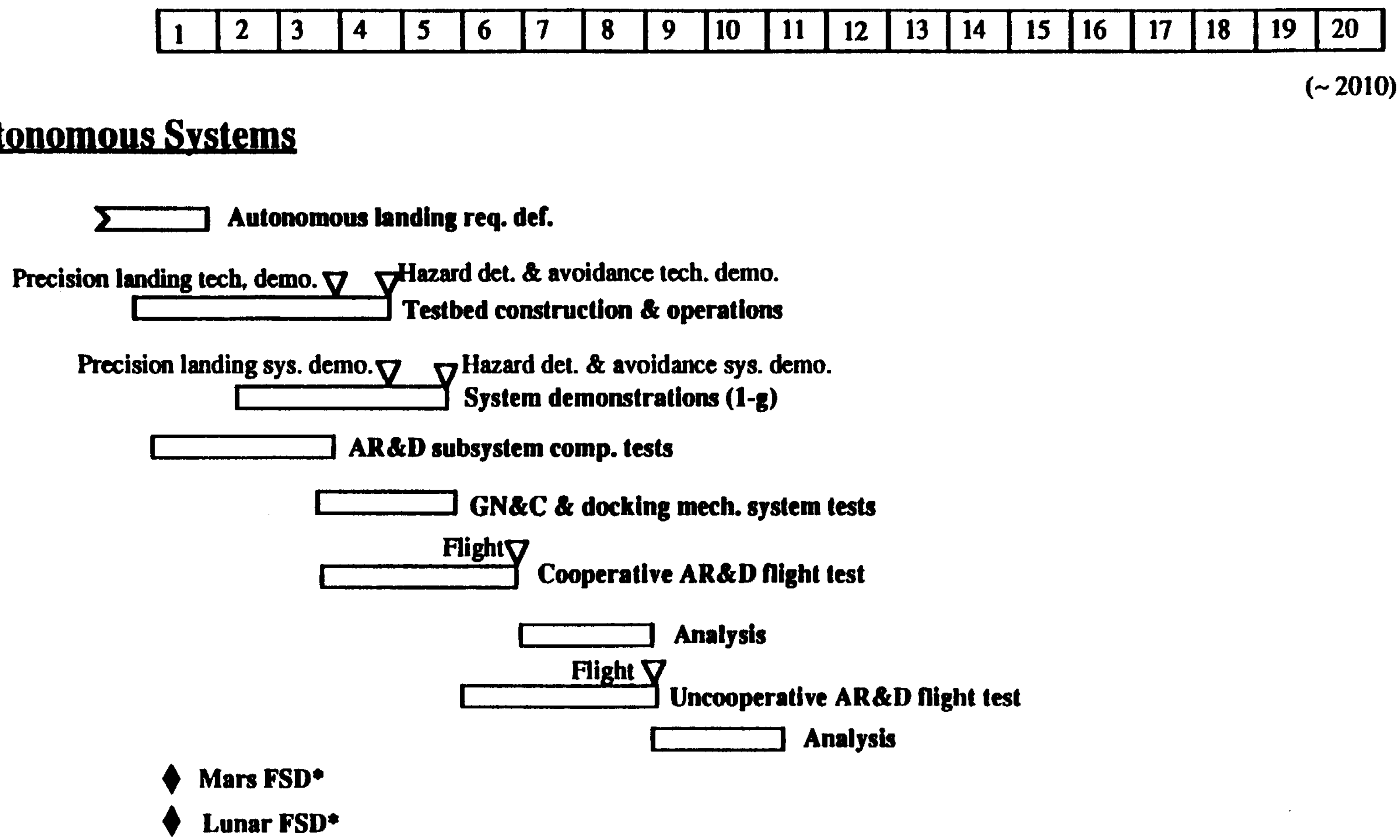
Preliminary SEI Technology Development Schedules (Cont.)

Autonomous Systems



* Technology should not present FSD threatening problems; current technologies adequate for minimum mission.

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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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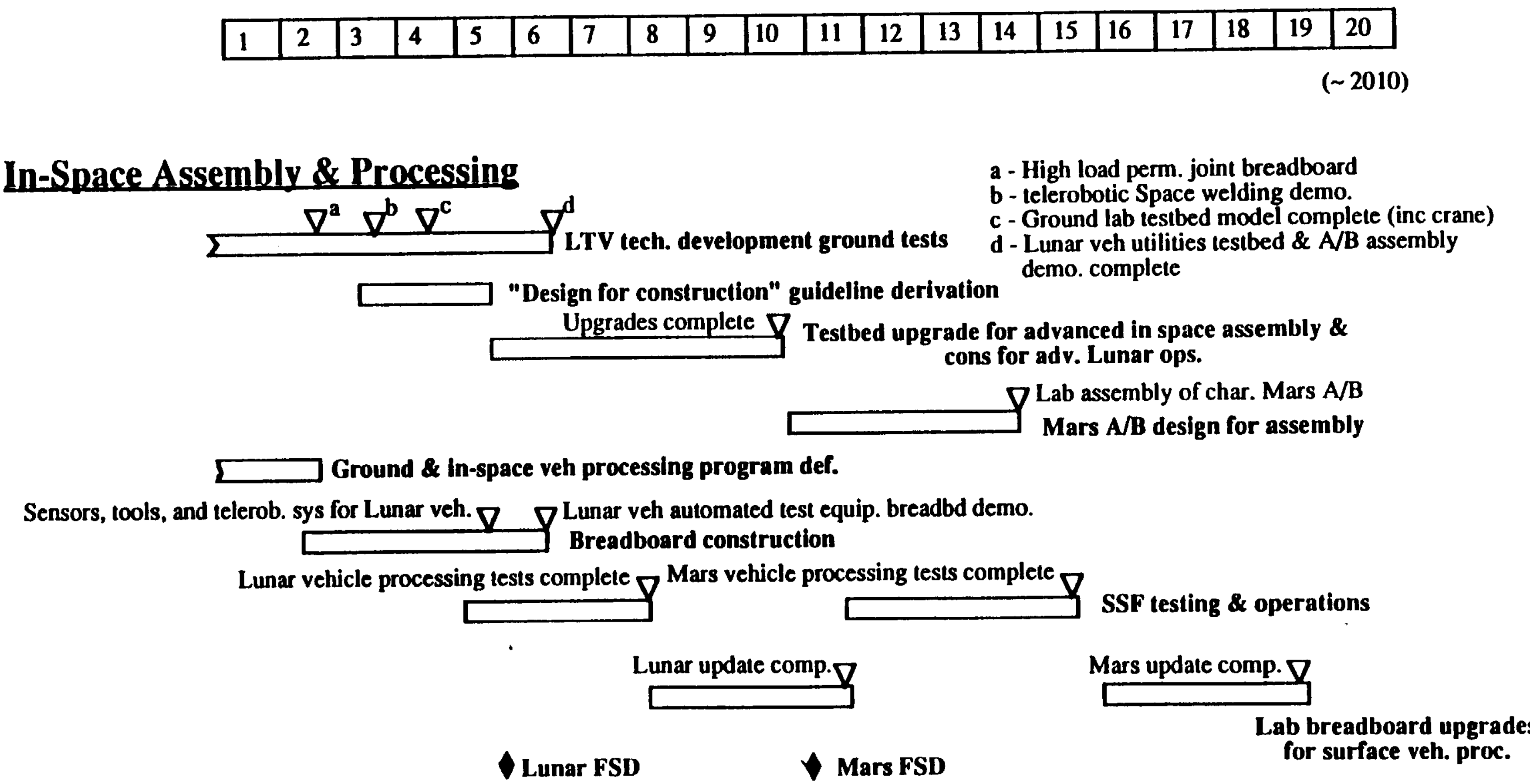

Preliminary SEI Technology Development Schedules (Cont.)

ADVANCED CIVIL SPACE SYSTEMS

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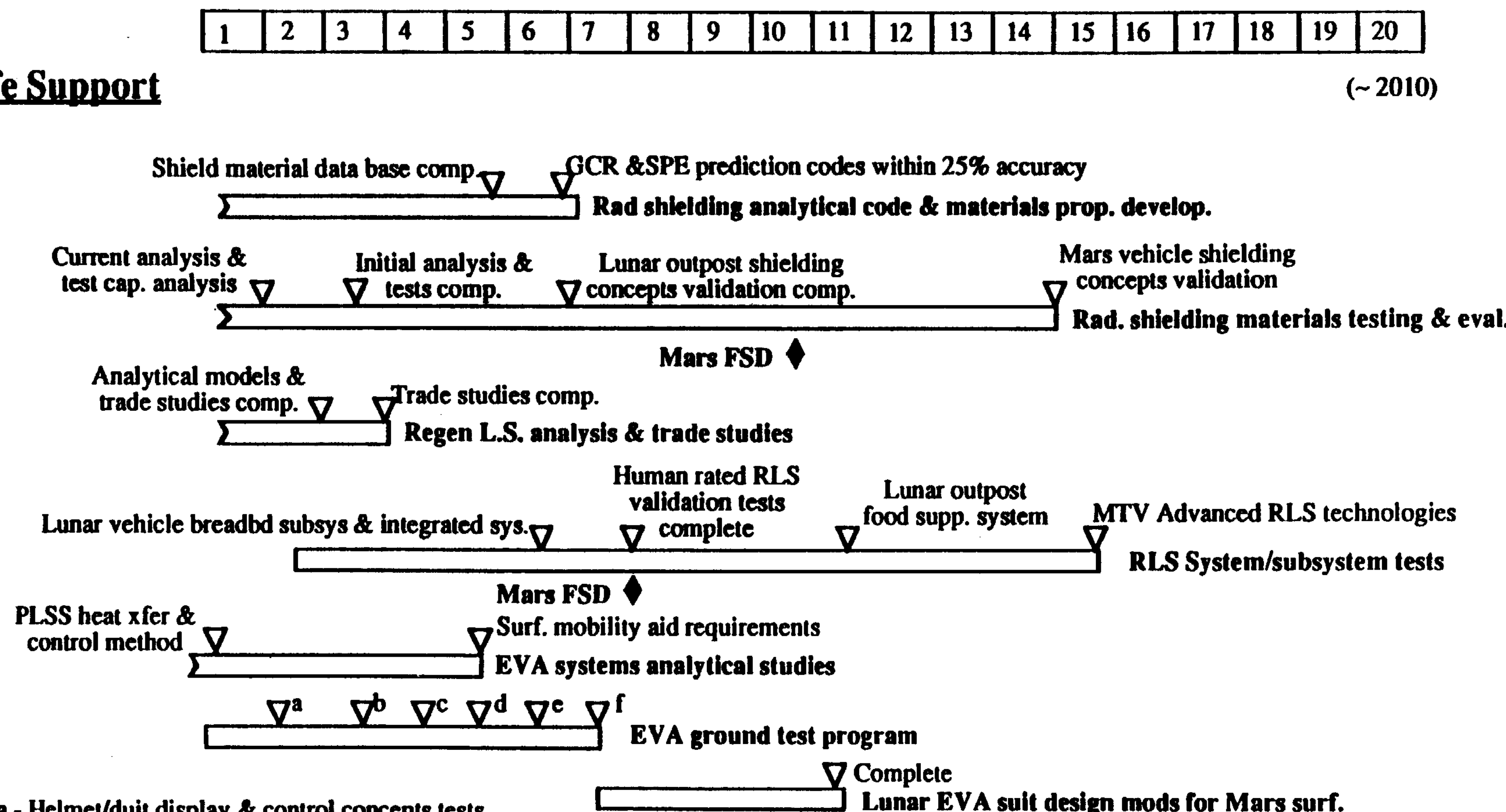
Life Support



a - Helmet/duit display & control concepts tests
b - Lunar surf suit breadbd test
c - Gloves & displays in simul. SSF environment
d - Regen. PLSS breadbd for lunar surf.
e - Verif tests of adv. dexterious gloves & disp.
f - Complete breadbd lunar EVA suit / simulated surf. cond.

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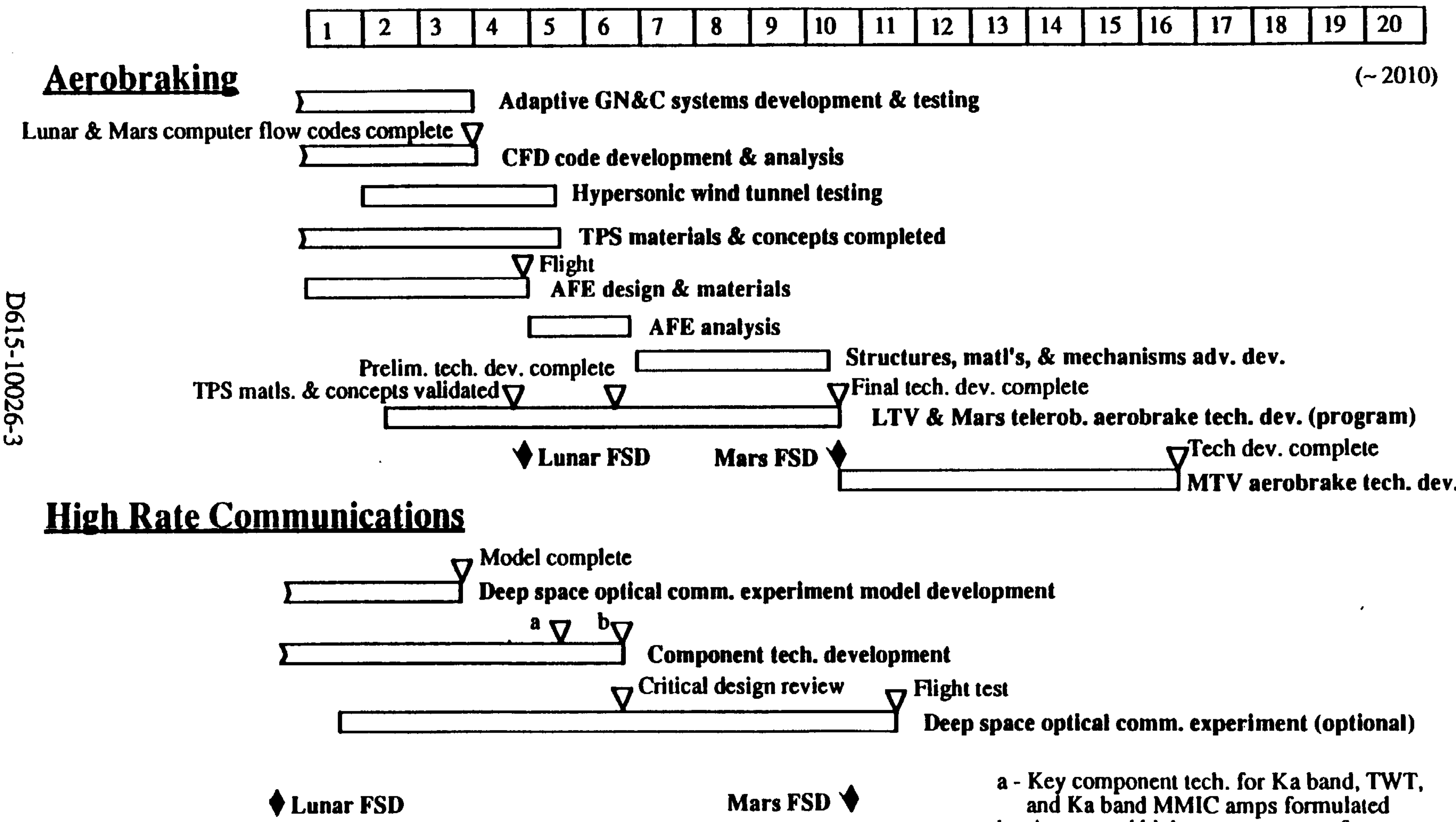
ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

BOEING



a - Key component tech. for Ka band, TWT, and Ka band MMIC amps formulated
b - Automated high rate comm ops for Lunar outpost & Mars robotic demo.

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Facilities

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Facilities

The facility needs have only been identified in this study; the extent of the impact is yet to be determined. A "bona fide" facility development plan has not been done as some of the requirements are only at a top-level needs evaluation. Therefore, the exact nature of the subsystems and their support facilities are undetermined. When these determinations have been made for the final NASA selected vehicle, the results must be integrated with the vehicle development schedule.

In addition to the information here, additional facility and equipment detail is shown in Ground subsection of the Support Systems section of this text. A current listing of the additional required facilities and equipment is shown in the "Special Ground and On-Orbit Processing Facility and Equipment Requirements" chart for processing the advanced vehicles. These requirements will impact the volumes shown for assembly, storage, and launch processing in the "Facilities Requirements" chart as well as the processing time shown in the "Assembly Time per Mission" chart. The information there is for the baseline Cryo/Aerobrake vehicle. All impacts will be to increase the processing time and working volumes required. Any facility requirements must be viewed in the light of and incorporated into the National Launch Facility Plan.

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Special Ground and On-Orbit Processing Facility and Equipment Requirements

Facilities/Equipment	NTR	NEP	SEP
Ground			
• Reactor/engine mating and processing facility	X	X	
• Nuclear fuel loading facility	X	X	
• Contaminated materials storage and disposal facility	X	X	
• Solar array/radiator packing and storage facility	X	X	X
• Alkali metals materials and transferring facility		X	
• Radiation/hazardous materials contamination treatment facility	X	X	
• Robotics to handle radioactive fuels and hazardous chemicals/materials and components	X	X	
• Vehicle truss processing and packaging facility	X	X	X
On-Orbit			
• On-orbit robotic welding and certification equipment	X	X	
• On-orbit alkali metal heating capability		X	
• On-orbit robotic repair/maintenance equipment	X	X	X

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Facility Requirements

	Assembly Volume	Storage Volume	Launch Processing
1	20694.13	0	0
2	20694.13	0	0
3	42233.11	0	0
4	56989.01	0	0
5	69879.77	10129.05	0
6	54623.87	10129.05	0
7	39222.88	25031.66	4626.85
8	39222.88	25031.66	0
9	49351.93	14902.61	0
10	20694.13	25031.66	18528.75
11	20694.13	34296.04	0
12	20694.13	34296.04	0
13	20694.13	25031.66	9264.38
14	39481.26	25031.66	0
15	39481.26	25031.66	0
16	0	25031.66	16912.13
17	18528.75	25031.66	0
18	18528.75	10129.05	0
19	0	25031.66	18528.75
20	0	34296.04	0
21	0	34296.04	0
22	0	25031.66	9264.38
23	0	25031.66	0
24	0	25031.66	0
25	0	10129.05	14902.61
26	21207.95	10129.05	0
27	21207.95	30387.15	0
28	0	30387.15	21207.95
29	0	30387.15	10129.05
30	0	30387.15	10129.05
31	0	20258.1	10129.05
32	0	20258.1	10129.05
33	0	20258.1	10129.05
34	0	20258.1	10129.05
35	0	10129.05	10129.05
36	0	10129.05	10129.05

Assembly Time per Mission

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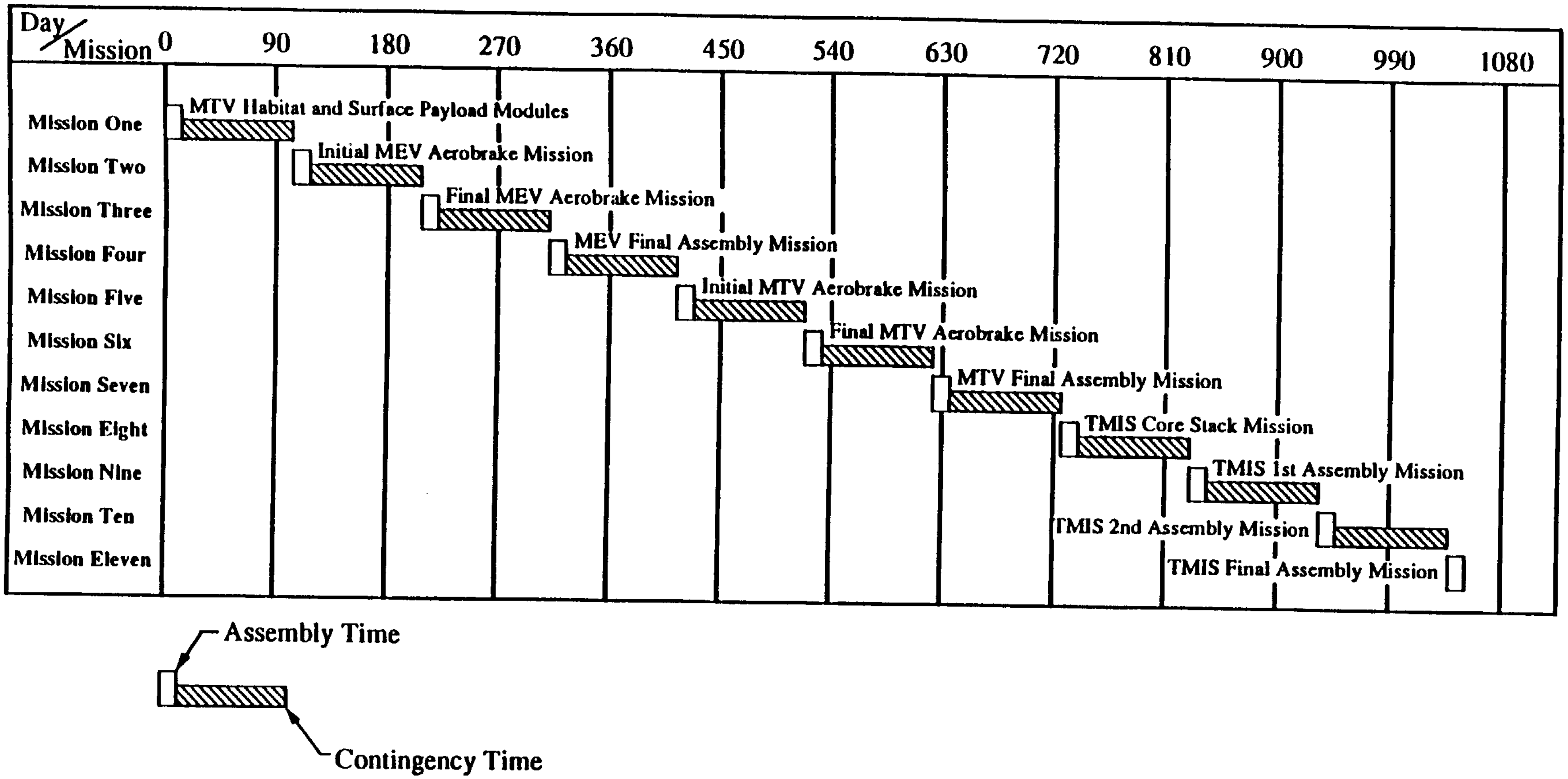

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Costs

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Programmatics

The objectives of the Programmatics task during the current phase of the study were: (1) realistic initial schedules that include initial critical path program elements; (2) initial descriptions of new or unique facilities requirements; (3) development of a stable, clear, responsive work breakdown structure (WBS) and WBS dictionary; (4) initial realistic estimates of vehicle, mission and program costs, cost uncertainties, and funding profile requirements; (5) initial risk analysis, and (6) early and continuing infusion of programmatics data into other study tasks to drive requirements/design/trade decisions.

The issues addressed during the study to date included: (1) capturing all potential long-lead program items such as precursor missions, technology advancement and advanced development, related infrastructure development, support systems and new or modified facility construction, since these are as important as cost and funding in assessing goal achievability; (2) incorporating sufficient operating margin in schedules to obtain high probability of making the relatively brief Mars launch windows; (3) the work breakdown structure must support key study goals such as commonality and (4) cost estimating accuracy and uncertainty are recurring issues in concept definition studies.

Introduction

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown in the "Overall Study Flow" chart. After the "neckdown" was completed, significant effort was put into programmatics.

As was indicated earlier, we established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for Lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in space transportation technologies as baselines for greater activity levels. The high level schedules developed for these three levels of activity are shown in the "Minimum Program, "Full Science Program" and "Industrialization and Settlement Program" charts and a comparison of them for both Lunar and Mars is shown in the "Lunar Program Comparison" and "Mars Program Comparison" charts.

Schedule/Network Development Methodology

A PC system called Open Plan by WST Corporation was used, which allows direct control and lower cost over a larger (mainframe) system. The network was purposely kept simple. Summary activities were used in development of the networks. When detailed to a lower level, some activities will require a different calendar than we used. One calendar with a five day work week - no holiday was used. Utilizing multicalendars on a summary network could confuse the development. The Preliminary WBS Structure Level 7 was followed for selection of work to be detailed. An example of Level 7 is: MEV Ascent Vehicle Structure/Mechanisms. We then developed a generic logic string of activities with standard durations for like activities. This logic was then applied against each WBS Level 7 element. To establish interface ties between logic strings and determination of major events, we used the Upper Level Summary Schedule and Summary Level Technology Schedule.

Goals/Purpose

There were two goals for the schedule/network development. These were:

a. Guidelines for Future Development. The schedules are a preliminary road map to follow in the development program.

b. Layout Basis Framework for Network. The networks can be used for future detail network development. This development can be in phases retaining unattended logic for areas which can be be detailed.

Status

Six preliminary networks have been developed. They are:

- Lunar minimum
- Lunar full science
- Lunar industrialization
- Mars missions
- Mars full science
- Mars settlement

These networks will be further developed as information becomes available The technology development plan schedules are shown in the Schedules section of this text; an example of the standard 6 year program phase C/D schedule is shown in the "Reference 6 yr.. Full Scale Development Program" chart. The network schedules developed during the study are available in the Final Report Cost Data Book.

Facilities

The facility requirements and approaches are discussed in the Facilities section of this text.

Development Implementation

The integrated technology advancement and full-scale development schedules for the NTR is shown in the "NTR Development Program". The MEV is developed according to the above mentioned standard 6-year FSD schedule. The Man-rating schedules for critical systems, that must be accomplished before first flight, are given in the next several man-rating charts. The long-duration Mars Tansit Habitat, and its critical subsystems, will require operational testing in space to qualify for the Mars mission. How all development and testing is actually done depends on program interrelationships between lunar and Mars missions.

Work Breakdown Structure

The approach to developing a WBS tree and dictionary was to use the Space Station Freedom Work Package One WBS as a point of departure to capture commonality, modularity and evolution potentials. We worked with MSFC to evolve the WBS illustrated in the six WBS charts given in this section. The WBS dictionary details are provided with the WBS tree in a separate deliverable document.

Cost Data

Overall Approach

Space transfer concept cost estimates were developed through parametric and detail estimating techniques using program/scenario plans and hardware and software descriptions combined with NASA and subcontractor data. Our estimating approach simulates the aerospace development and production environment. It also reflects program options not typical of aerospace programs. This flexibility allows assessment of innovative program planning concepts.

Several tools were employed in this analysis. For developing estimates the Boeing Parametric Cost Model (PCM) designed specifically for advanced system estimating was used. It utilizes a company-wide, uniform computerized data base containing historical data compiled since 1969. The second major tool is a Boeing developed Life Cycle Cost Model. The third tool is the Boeing developed Return on Investment (ROI) Analyses.

The approach to cost estimating was to use the PCM to establish DDT&E and manufacturing cost of major hardware components or to use other estimates, (e.g. Nuclear Working Group estimator) if they were considered superior and then feed them to the LCC model. Variations on equipment hardware or mission alternatives can be run through the LCC and then compared for a return on investment. This flow is illustrated in t "Costing Methodology Flow" chart. We were able to investigate alternative concepts quickly, giving system designers more data for evolving scenario/mission responsive concepts. Transportation concepts, trade studies, and "neckdown" efforts were supported by this approach.

Parametric Cost Model

PCM develops cost from the subsystem level and builds upward to obtain total program cost. Costs are estimated from physical hardware descriptions (e.g., weights and complexities) and program parameters (e.g., quantities, learning curves, and integration levels). Known costs are input directly into the estimate when available; the model assesses the necessary system engineering and system test efforts needed for integration into the program. The PCM working unit is man-hours, which allows relationships that tie physical hardware descriptions first to design engineering or basic factory labor, and then through the organizational structure to pick up functional areas such as systems engineering, test, and development shop. Using man-hours instead of dollars for estimating relationships enables more reliable estimates. The PCM features, main inputs, and results are shown in the "Boeing Parametric Cost Model (PCM)" chart. The applicable PCM results, in constant 1990 dollars, are then put into the Life Cycle Cost Model to obtain cost spreads for the various missions/programs. The various hardware components costed for the three different missions/programs are shown in the "LCCM Hardware Assignment " chart.

The development of space hardware and components needed to accomplish the three different Lunar/Mars missions were identified. These components are grouped into three different categories defined below.

HLLV(Heavy Lift Launch Vehicle) is the booster required to lift personnel, cargo and fuels into LEO and support the LEO node operations.

Propulsion Includes the space propulsion system required to transfer people, cargo and equipment out of LEO and into space. Space means Lunar, Mars and Earth destinations. Propulsion Systems also include an all-propulsive cryogenic Trans Mars Injection System (TMIS) for the Minimum Mission, the Nuclear Electric Propulsion Stage for the Settlement/Industrial Missions.

Modules Include the space systems that are required to transfer people, cargo and equipment from LEO to Lunar and Mars orbit; to de-orbit and sustain life and operations on the Lunar and Mars Surface; and, finally, to return personnel and equipment to LEO.

Cost Buildups

The PCM cost Model can be used directly to obtain complete DDT&E cost, including production of major test articles, by entering into the manufacturing section the equivalent numbers of units for each item, including the first flight article. However, when operated in this way, PCM does not give the first unit cost. To save time, we operated PCM so as to give first unit cost, which we needed for life cycle cost analyses, and used the first unit cost to manually estimate the test hardware content of the DDT&E program. The "wrap factors" shown in the cost buildup sheets were derived from the PCM runs as the factor that is applied to design engineering cost to obtain complete design and development costs, e.g. including non-recurring items such as systems engineering and tooling development.

Life Cycle Cost Model

The LCCM cost data is a composite of HLLV costs, launch base facilities cost estimate based on $/sq. ft. and parametric estimates derived from the Parametric Cost Model. The principal source of information is from the PCM. All hardware cost estimates, with the exception of HLLV, have been developed with this model.

The LCCM consists of three individual models. One model is for the Minimum Program Scale; the second is for the Full Science Program Scale; while the third model is for the Settlement/Industrialization Program Scale. The Minimum Program meets the President's Space Exploration Initiative (SEI) objectives. These capabilities include permanent Lunar facilities but not permanent human presence and three missions to Mars. The Full Science program not only meets the President's SEI objectives but also provides for long term bases for far-ranging surface exploration. The Settlement/Industrialization program accomplishes the objectives of the Minimum and Full Science program scales and additionally returns practical benefits to Earth. These models were developed using the three architecture levels described in the Boeing manifest worksheets. Total cost for each system are tabulated by year and each year's totals feed into a summary sheet that calculates the total program cost for each level. Since the LCCM results are mission related, not just vehicle related, they are not provided here but are available in the Final Report Cost Data Book. The LCCM was developed using Microsoft Excel version 2.2 for the Macintosh computer. Any Macintosh equipped with Excel 2.2 can be used to execute the model.

Return On Investment

One of the principal uses of the LCCM is to develop trades and return on investment for technology options. As shown in the " Costing Methodology Flow" chart, two separate life cycle cost models (which include DDT&E and production cost data derived from the parametric cost models) must be developed for each ROI case; a reference, and a case utilizing a technology option. The two life cycle cost streams are separately entered, and the ROI model is executed. The flow also illustrates that not all of the data entered into the life cycle cost model is derived from available costing software. Technical analysis must accompany this data. For example, the number of units which must be produced for the DDT&E program must be determined. This is done at the subsystem level based on knowledge of past programs, and proposed system/subsystem tests. Since the ROI analysis is mission related, not just vehicle related, the data is not presented here but is available in the Final Report Costing Data Book.

Results

A summary of the cost data produced by the PCM for the NTR vehicle are given in the "Mars NTR PCM Summary" and "Mars NTR PCM Summary - continued" charts. The

PCM program was used to produce DDT&E and production cost estimates for each of our reference Mars and lunar vehicles to the subsystem level. The DDT&E costs generated by the PCM do not include all of the necessary hardware for the first mission vehicle. Hence all necessary additional units (prototypes, test units, lab units, etc.) were added into the vehicle cost buildups as shown in the "NTR Cost Buildup" charts . The total DDT&E includes additional costs (e.g.. additional units in the DDT&E program), contractor fees and the engineering wrap factor. The total DDT&E from the cost buildup and the unit cost from the PCM are the primary vehicle cost inputs to the LCC model.

Risk Analyses

Risk analyses were conducted to develop an initial risk assessment for the various architectures. This presentation of risk analysis results considers development risk, man-rating requirements, and several aspects of mission and operations risk.

Development Risk

All of the architectures and technologies investigated in this study incur some degree of development risk; none are comprised entirely of fully developed technology. Development risks are correlated directly with technological uncertainties. We identified the following principal risks:

Cryogenics - High-performance insulation systems involve a great many layers of multi-layer insulation (MLI), and one or more vapor-cooled shields. Analyses and experiments have indicated the efficacy of these, but demonstration that such insulation systems can be fabricated at light weight, capable of surviving launch g and acoustics loads, remains to be accomplished. In addition, there are issues associated with propellant transfer and zero-g gauging. These, however, can be avoided for early lunar systems by proper choice of configuration and operations, e.g. the tandem-direct system recommended elsewhere in this report. This presents the opportunity to evolve these technologies with operations of initial flight systems.

Engines - There is little risk of being able to provide some sort of cryogenic engine for lunar and Mars missions. The RL- 10 could be modified to serve with little risk; deep throttling of this engine has already been demonstrated on the test stand. The risk of developing more advanced engines is also minimal. An advanced development program in this area serves mainly to reduce development cost by pioneering the critical features prior to full-scale development.

Aerocapture and aerobraking - There are six potential functions, given here in approximate ascending order of development risk: aero descent and landing of crew capsules returning from the Moon, aerocapture to low Earth orbit of returning reusable lunar vehicles, landing of Mars excursion vehicles from Mars orbit, aero descent and landing of crew capsules returning from Mars, aerocapture to low Earth orbit of returning Mars vehicles, and aerocapture to Mars orbit of Mars excursion and Mars transfer vehicles. The "Development Risk Assessment for Aerobraking by Function" chart provides a qualitative development risk comparison for these six functions.

Aerocapture of vehicles requires large aerobrakes. For these to be efficient, low mass per unit area is required, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials. By comparison, the crew capsules benefit much less from high performance structures and TPS.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk that has not yet been solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both. While the design challenge can probably be met, aerobrake assembly is a difficult design and development challenge, representing an important area of risk.

Nuclear thermal rockets - The basic technology of nuclear thermal rockets was developed and demonstrated during the 1960s and early 1970s. The development risk to reproduce this technology is minimal, except in testing as described below. Current studies are recommending advances in engine performance, both in specific impulse (higher reactor temperature) and in thrust-to-weight ratio (higher reactor power density). The risks in achieving these are modest inasmuch as performance targets can be adjusted to technology performance.

Reactor and engine tests during the 1960s jetted hot, slightly radioactive hydrogen directly into the atmosphere. Stricter environmental controls since that time prohibit discharge of nuclear engine effluent into the atmosphere. Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines. Full- containment facilities will be required to contain all the hydrogen effluent, presumably oxidize it to water, and remove the radioactivity.

Electric Propulsion Power Management and Thrusters - Power management and thrusters are common to any electric propulsion power source (nuclear, solar, or beamed power). Unique power management development needs for electric propulsion are (1) minimum mass and long life, (2) high power compared to space experience, i.e. megawatts instead of kilowatts, (3) fast arc suppression for protection of thrusters. Minimizing mass of power distribution leads to high distribution voltage and potential problems with plasma losses, arcing, and EMI. Thus while power management is a mature technology, the unique requirements of electric propulsion introduce a number of development risks beyond those usually experienced in space power systems.

Electric thruster technology has been under development since the beginning of the space program. Small thrusters are now operational, such as the resistance-heat-augmented hydrazine thrusters on certain communications spacecraft. Small arc and ion thrusters are nearing operational use for satellite stationkeeping.

Space transfer demands on electric propulsion performance place a premium on high power in the jet per unit mass of electric propulsion system. This in turn places a premium on thruster efficiency; power in the jet, not electrical power, propels spaceships. Space transfer electric propulsion also requires specific impulse in the range 5000 to 10,000 seconds. Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver this performance. Ion thrusters have acceptable efficiency but relatively low power per unit of ion beam emitting area. MPD thruster technology can deliver the needed Isp with high power per thruster, but has not yet reached efficiencies of interest. Circular ion thrusters have been built up to 50 cm diameter, with spherical segment ion beam grids. These can absorb on the order of 50 kWe each. A 10 MWe system would need 200 operating thrusters. The development alternatives all have significant risk: (1) Advance the state of the art of MPD thrusters to achieve high efficiency; (2) Develop propulsion systems with large numbers of thrusters and control systems; or (3) Advance the state of the art of ion thrusters to much larger size per thruster.

Nuclear power for electric propulsion - Space power reactor technology now under development (SP-100) may be adequate; needed advances are modest. Advanced power conversion systems are required to obtain power-to-mass ratios of interest. The SP-100 baseline is thermoelectric, which has no hope of meeting propulsion system performance needs. The most likely candidates are the closed Brayton (gas) cycle and the potassium Rankine (liquid/vapor) cycle. (Potassium provides the best match of liquid/vapor fluid properties to desired cycle temperatures.) Stirling cycle, thermionics, and a high-temperature thermally-driven fuel cell are possibilities. The basic technology for Brayton and Rankine cycles are mature; both are in widespread industrial use. Prototype space power Brayton and Rankine turbines have run successfully for thousands of hours in laboratories. The development risk here is that these are very complex systems; there is no experience base for coupling a space power reactor to a dynamic power conversion cycle; there is no space power experience base at the power levels needed; and these systems, at power levels of interest for SEI space transfer application, are large enough to require in-space assembly and checkout. Space welding will be required for fluid systems assembly.

Solar power for space transfer propulsion - Solar power systems for space propulsion must attain much higher power-to-mass ratios than heretofore achieved. This implies a combination of advanced solar cells, probably multi-band-gap, and lightweight structural support systems. Required array areas are very large. Low-cost arrays, e.g. $100/watt, are necessary for affordable system costs, and automated construction of the large area structures, arrays, and power distribution systems appears also necessary. Where the nuclear electric systems are high development risk because of complexity and the lack of experience base at relevant power levels and with the space power conversion technologies, most of the solar power risk appears as technology advancement risk. If the technology advancements can be demonstrated, development risk appears moderate.

Avionics and software - Avionics and software requirements for space transfer systems are generally within the state of the art. New capability needs are mainly in the area of vehicle and subsystem health monitoring. This is in part an integration problem, but new techniques such as expert and neural systems are likely to play an important role.

An important factor in avionics and software development is that several vehicle elements having similar requirements will be developed, some concurrently. A major reduction in cost and integration risk for avionics can be achieved by advanced development of a "standard" avionics and software suite, from which all vehicle elements would depart.

Further significant cost savings are expected from advancements in software development methods and environments.

Environmental Control and Life Support (ECLS) - The main development risk in ECLS is for the Mars transfer habitat system. Other SEI space transfer systems have short enough operating durations that shuttle and Space Station Freedom ECLS system derivatives will be adequate. The Mars transfer requirement is for a highly closed physio-chemical system capable of 3 years' safe and dependable operation without resupply from Earth. The development risk arises from the necessity to demonstrate long life operation with high confidence; this may be expensive in cost and development schedule.

Man-Rating Approach

Man-rating includes three elements: (1) Design of systems to manned flight failure tolerance standards, (2) Qualification of subsystems according to normal man-rating requirements, and (3) Flight demonstration of critical performance capabilities and functions prior to placing crews at risk. Several briefing charts follow: the first summarizes a recommended approach and lists the subsystems and elements for which man-rating is needed; subsequent charts present recommended man-rating plans.

Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbital launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks.

Earth launch - The Earth launch risk to in-space transportation is the risk of losing a payload because of a launch failure. Assembly sequences are arranged to minimize the impact of a loss, and schedules include allowances for one make-up launch each mission opportunity.

Assembly and Orbital Launch Operations - Four sub-areas are covered: assembly, test and on-orbit checkout, debris, and inadvertent re-entry.

Assembly operations risk is reduced by verifying interfaces on the ground prior to launch of elements. Assembly operations equipment such as robot arms and manipulators will undergo space testing at the node to qualify critical capabilities and performance prior to initiating assembly operations on an actual vehicle.

Assembly risk varies widely with space transfer technology. Nuclear thermal rocket vehicles appear to pose minimum assembly risk; cryo/aerobraking are intermediate, and nuclear and solar electric systems pose the highest risk.

Test and on-orbit checkout must deal with consequences of test failures and equipment failures. This risk is difficult to quantify with the present state of knowledge. Indications are: (1) large space transfer systems will experience several failures or anomalies per day. Dealing with failures and anomalies must be a routine, not exceptional, part of the operations or the operations will not be able to launch space transfer systems from orbit; (2) vehicles must have highly capable self-test systems and must be designed for repair, remove and replace by robotics where possible and for ease of repair by people where robotics cannot do the job; (3) test and on-orbit checkout will run concurrently with propellant loading and launch countdowns. These cannot take place on Space Station Freedom. Since the most difficult part of the assembly, test and checkout job must take place off Space Station Freedom the rest of the job probably should also.

Orbital debris presents risk to on-orbit operations. Probabilities of collision are large for SEI-class space transfer systems in low Earth orbit for typical durations of a year or more. Shielding is mandatory. The shielding should be designed to be removed before orbital launch and used again on the next assembly project.

Creation of debris must also be dealt with. This means that (1) debris shielding should be designed to minimize creation of additional debris, especially particles of dangerous size, and (2) operations need to be rigorously controlled to prevent an inadvertent loss of tools and equipment that will become a debris hazard.

Inadvertent re-entry is a low but possible risk. Some of the systems, especially electric propulsion systems, can have very low ballistic coefficient and therefore rapid orbital decay rate. Any of the SEI space transfer systems will have moderately low ballistic coefficient when not loaded with propellant. While design details are not far enough along to make a quantitative assessment, parts of these vehicles would probably survive reentry to become ground impact hazards in case of inadvertent reentry. For nuclear systems, it will be necessary to provide special support systems and infrastructure to drive the probability of inadvertent reentry to extremely low levels.

Launch Windows - Launch windows for single-burn high-thrust departures from low Earth orbit are no more than a few days because regression of the parking orbit line of nodes causes relatively rapid misalignment of the orbit plane and departure vector. For lunar missions, windows recur at about 9-day intervals.

For Mars, the recurrence is less frequent, and the interplanetary window only lasts 30 to 60 days. It is important to enable Mars launch from orbit during the entire interplanetary window. Three-impulse Mars departures make this possible; a plane change at apogee of the intermediate parking orbit provides alignment with the departure vector. Further analysis of the three-burn scheme is needed to assess penalties and identify circumstances where it does not work.

Launch window problems are generally minimal for low-thrust (electric propulsion) systems.

Mission Risk - Comparative mission risk was analyzed by building risk trees and performing semi-quantitative analysis. The next chart presents a comparison of several mission modes; after that are the risk trees for these modes.

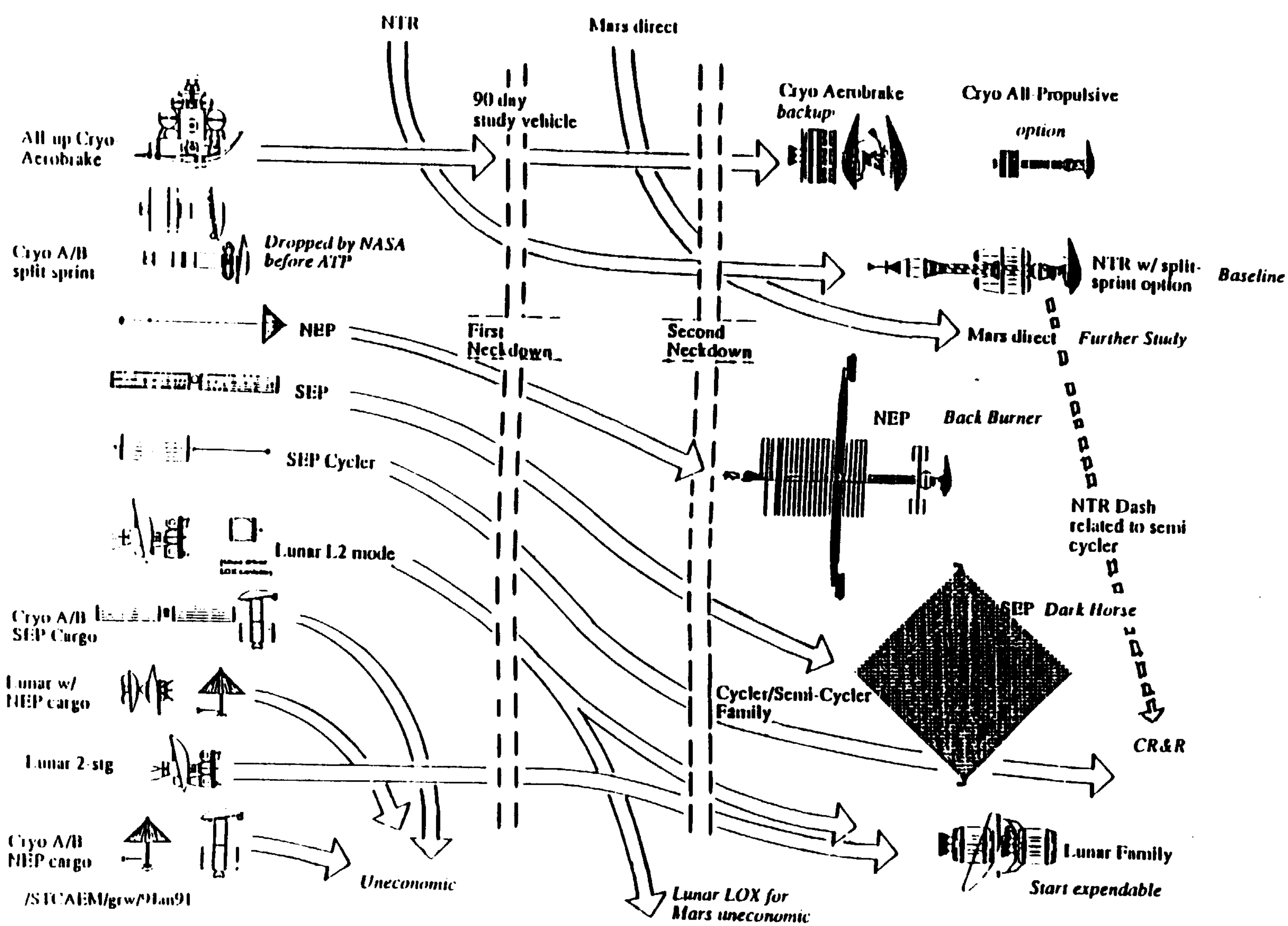
Ionizing Radiations and Zero G - The threat from ionizing radiations is presented elsewhere in this document. Presented here are the mitigating strategies for ionizing radiations and zero g.

Nuclear systems operations present little risk to flight crews. Studies by University of Texas at Austin showed that radiation dose to a space station crew from departing nuclear vehicles is very small provided that sensible launch and flight strategies are used. On-board crews are protected by suitable shielding and by arrangement of the vehicle, i.e. hardware and propellant between reactors and the crew and adequate separation distances. After nuclear engines are shut off, radiation levels drop rapidly so that maneuvers such as departure or return of a Mars excursion vehicle are not a problem. On-orbit operations around a returned nuclear vehicle are deferred until a month or two after shutdown, by which time radioactivity of the engine is greatly reduced.

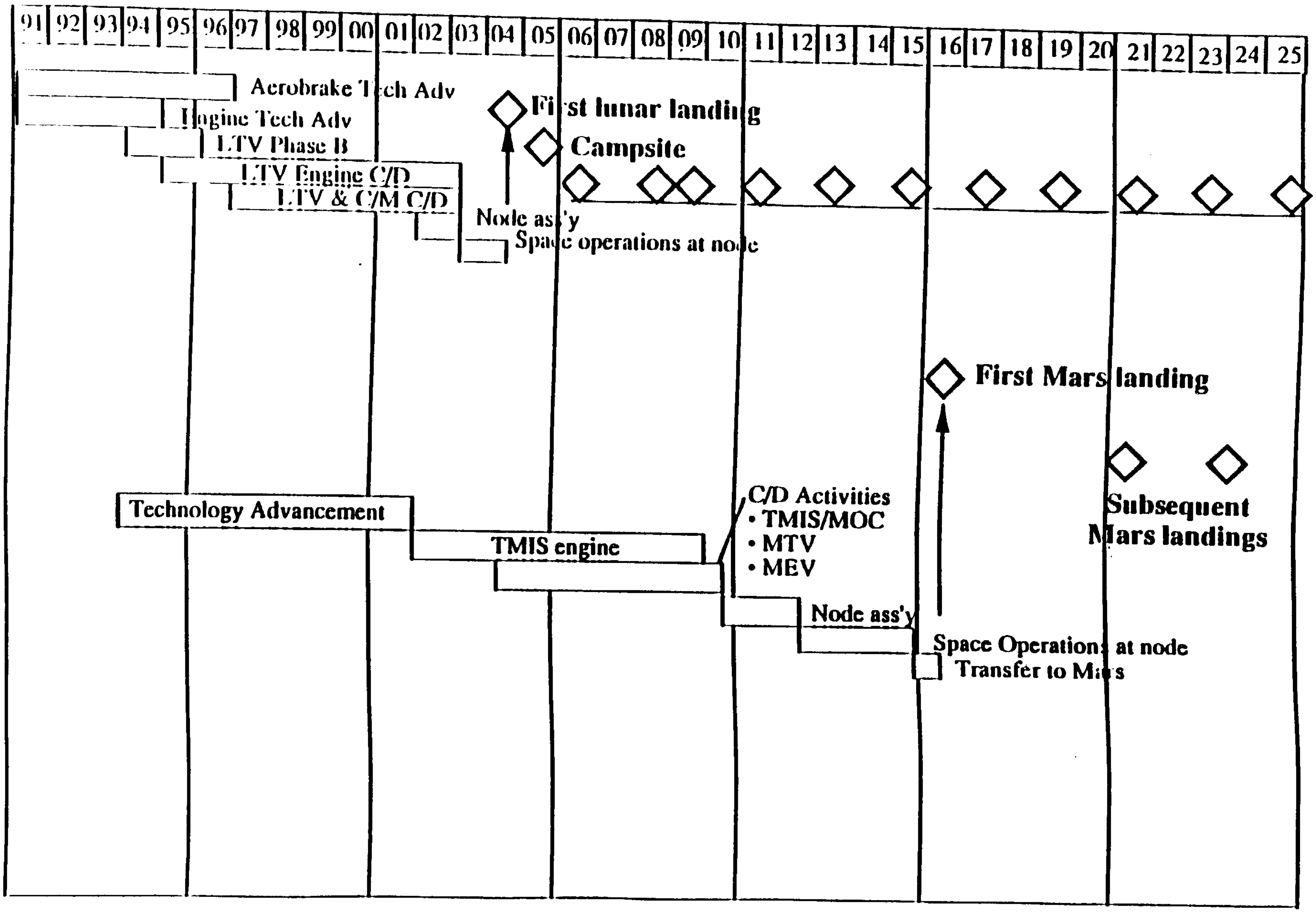
Reactor disposal has not been completely studied. Options include solar system escape and parking in stable heliocentric orbits between Earth and Venus.

Crew radiation dose abatement employs "storm shelters" for solar flares, and either added shielding of the entire vehicle or fast transfers (or both) to reduce galactic cosmic ray exposure. Assessments are in progress; tradeoffs of shielding versus fast trips have yet to be completed. Expected impact for lunar missions is negligible and for Mars missions, modest.

Overall Study Flow



Minimum Program



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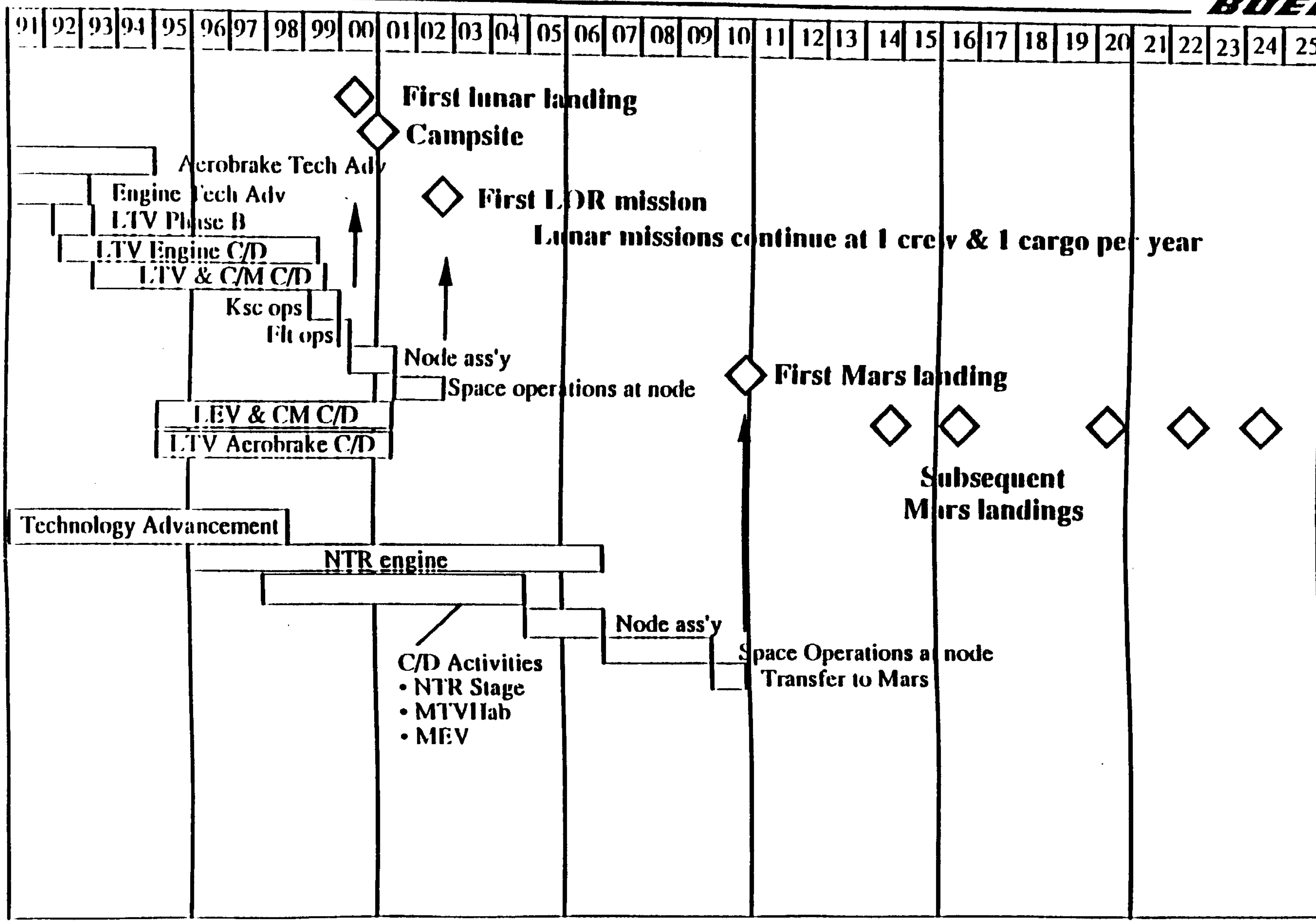
Full Science Program



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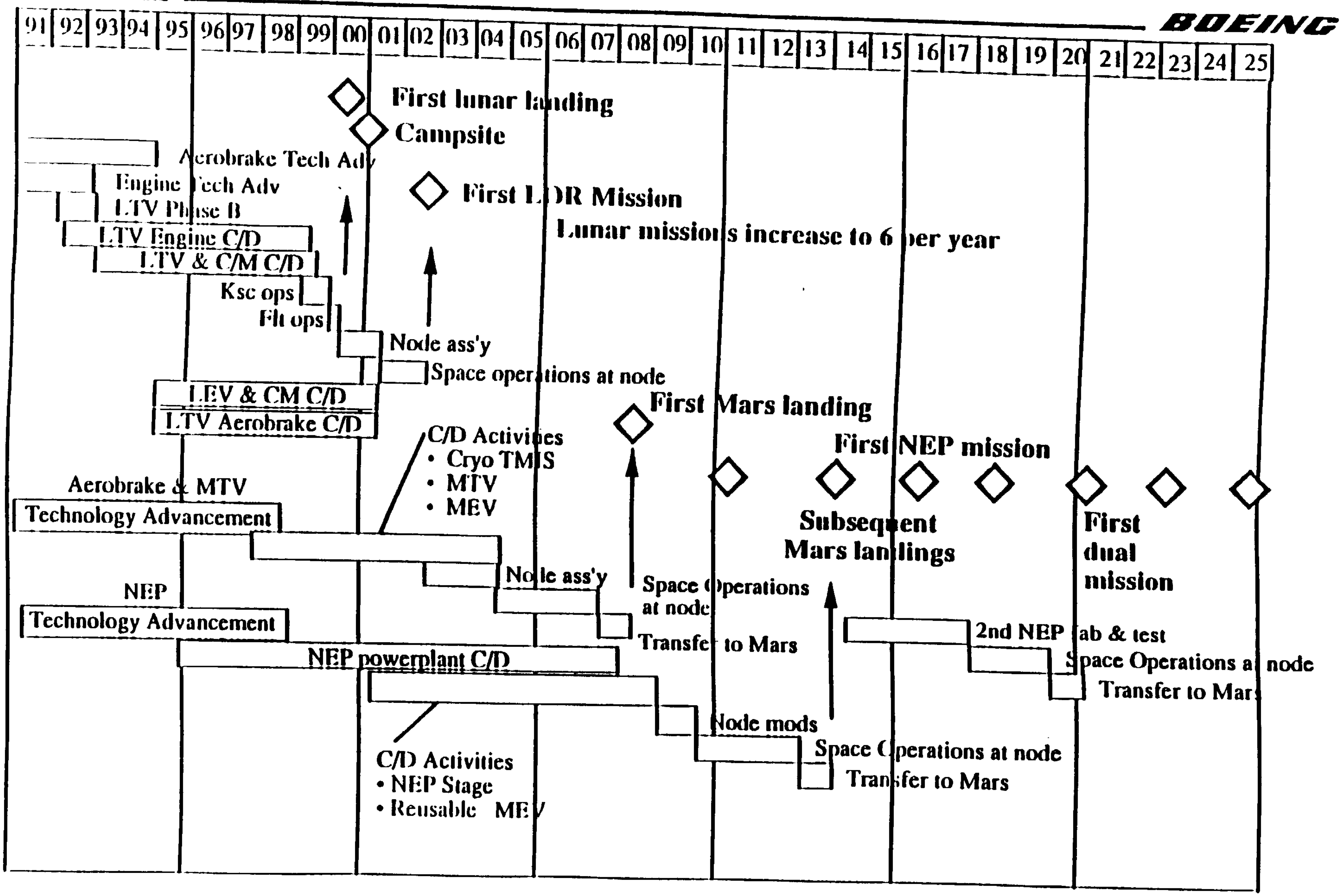
ADVANCED CIVIL SPACE SYSTEMS

Industrialization and Settlement Program

BOEING



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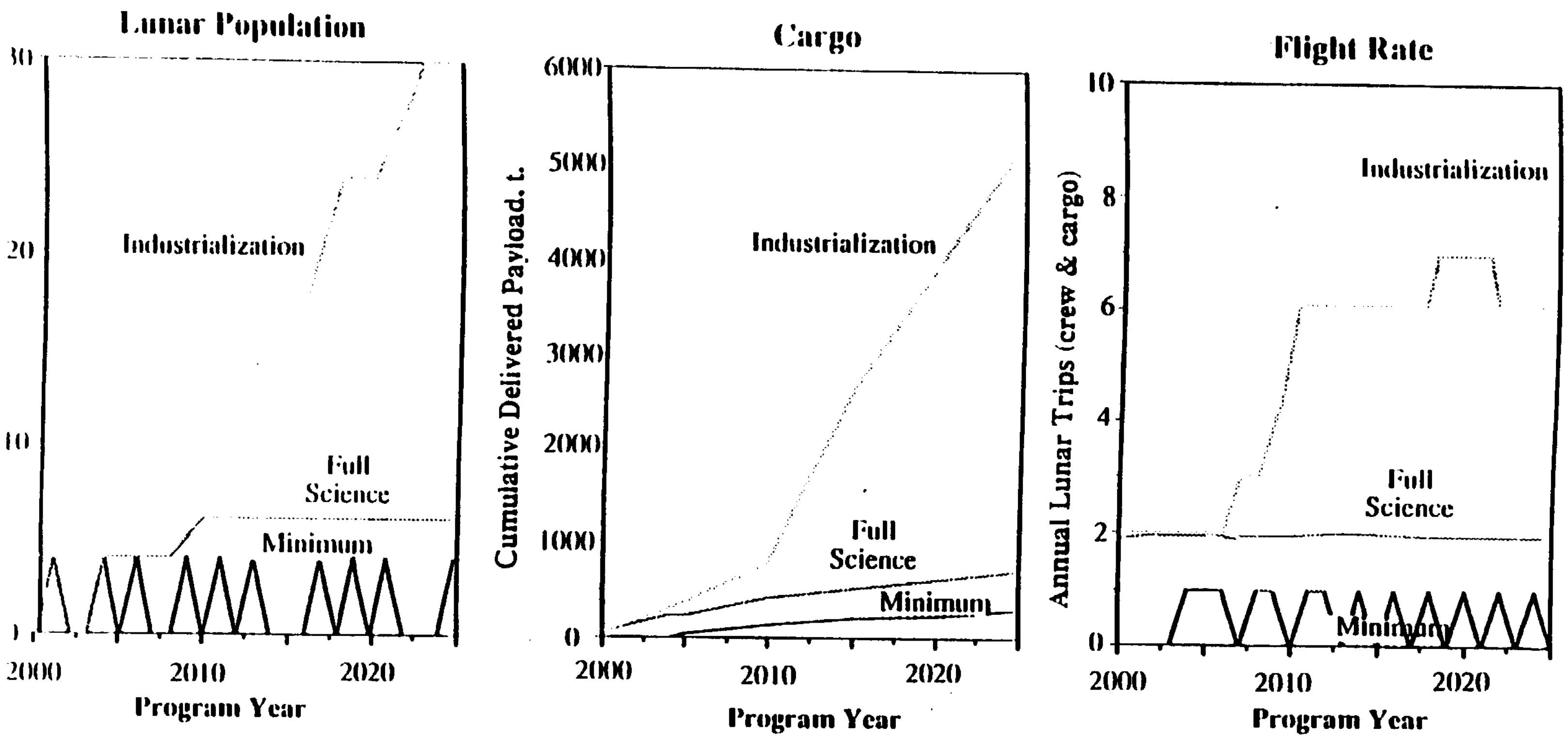
Lunar Program Comparison



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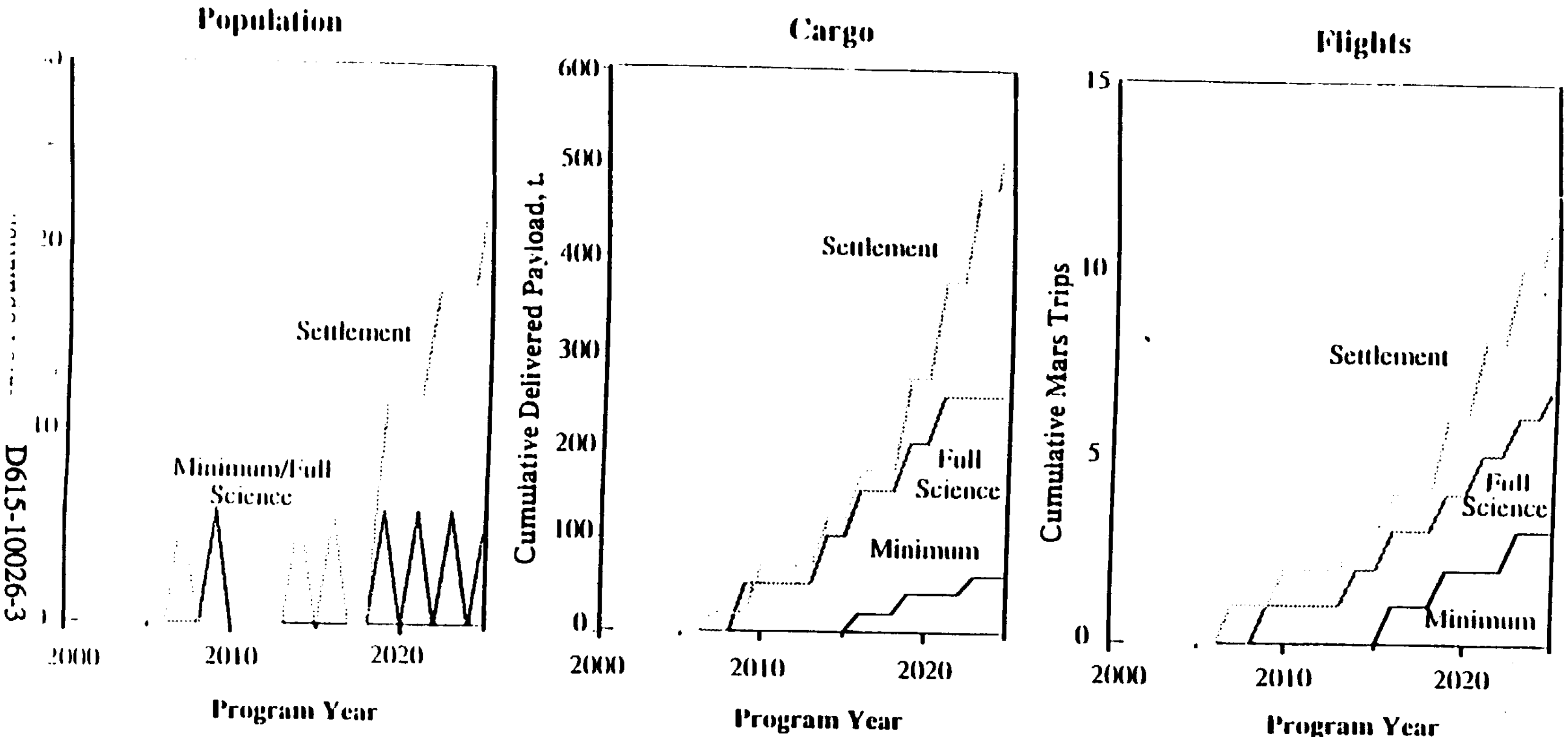
Mars Program Comparisons



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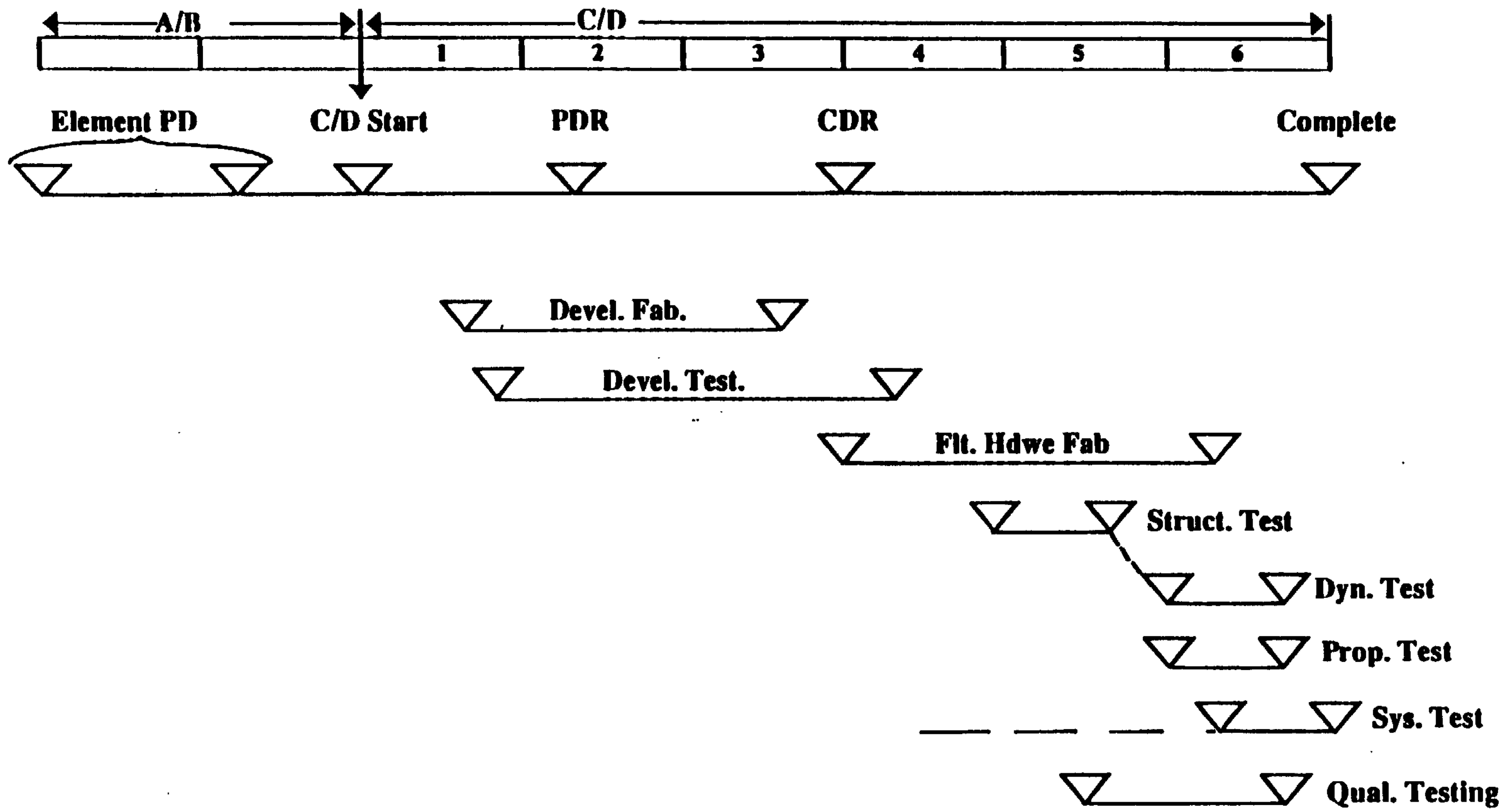
Reference 6 yr Full-Scale Development Schedule



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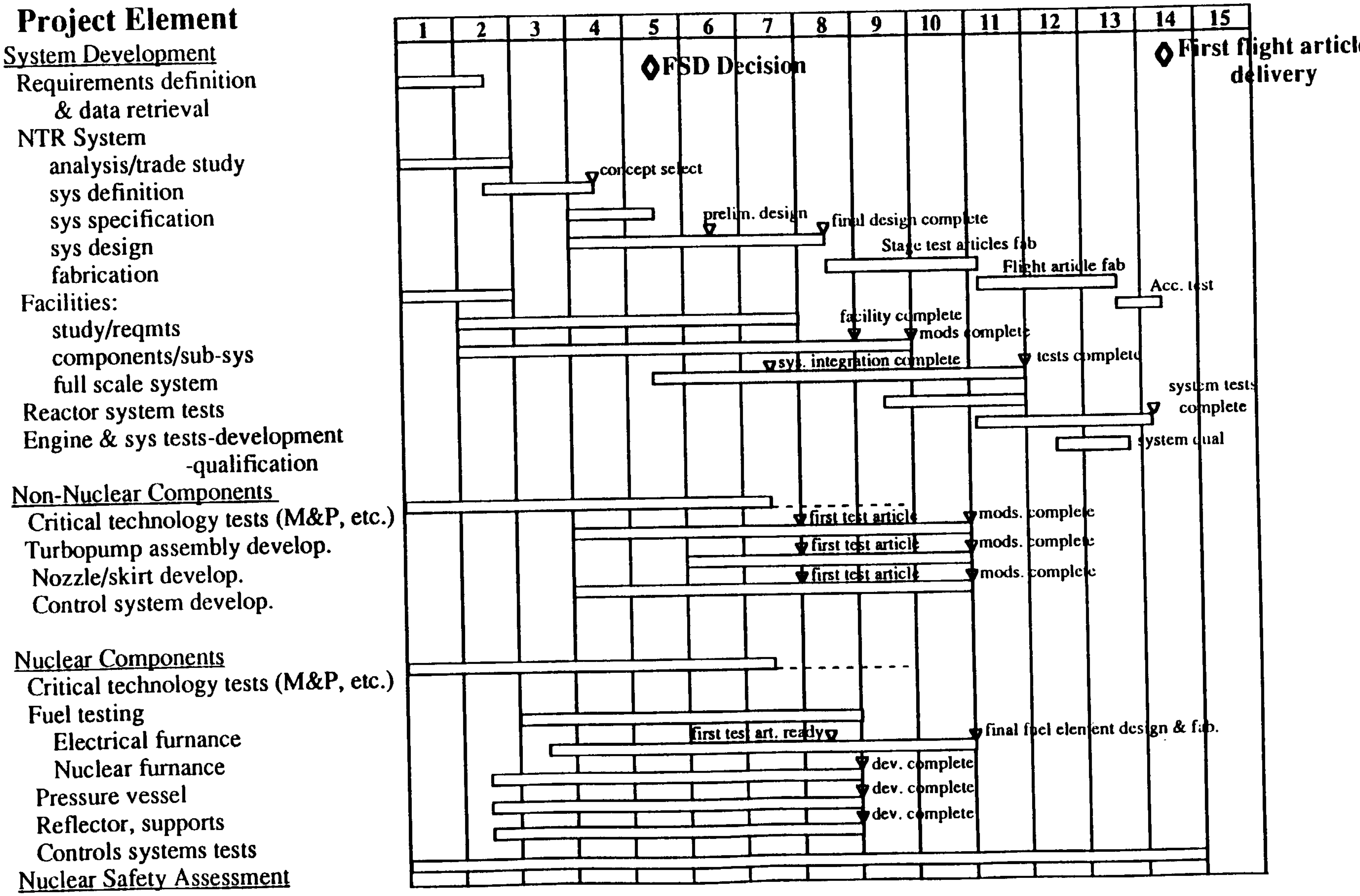
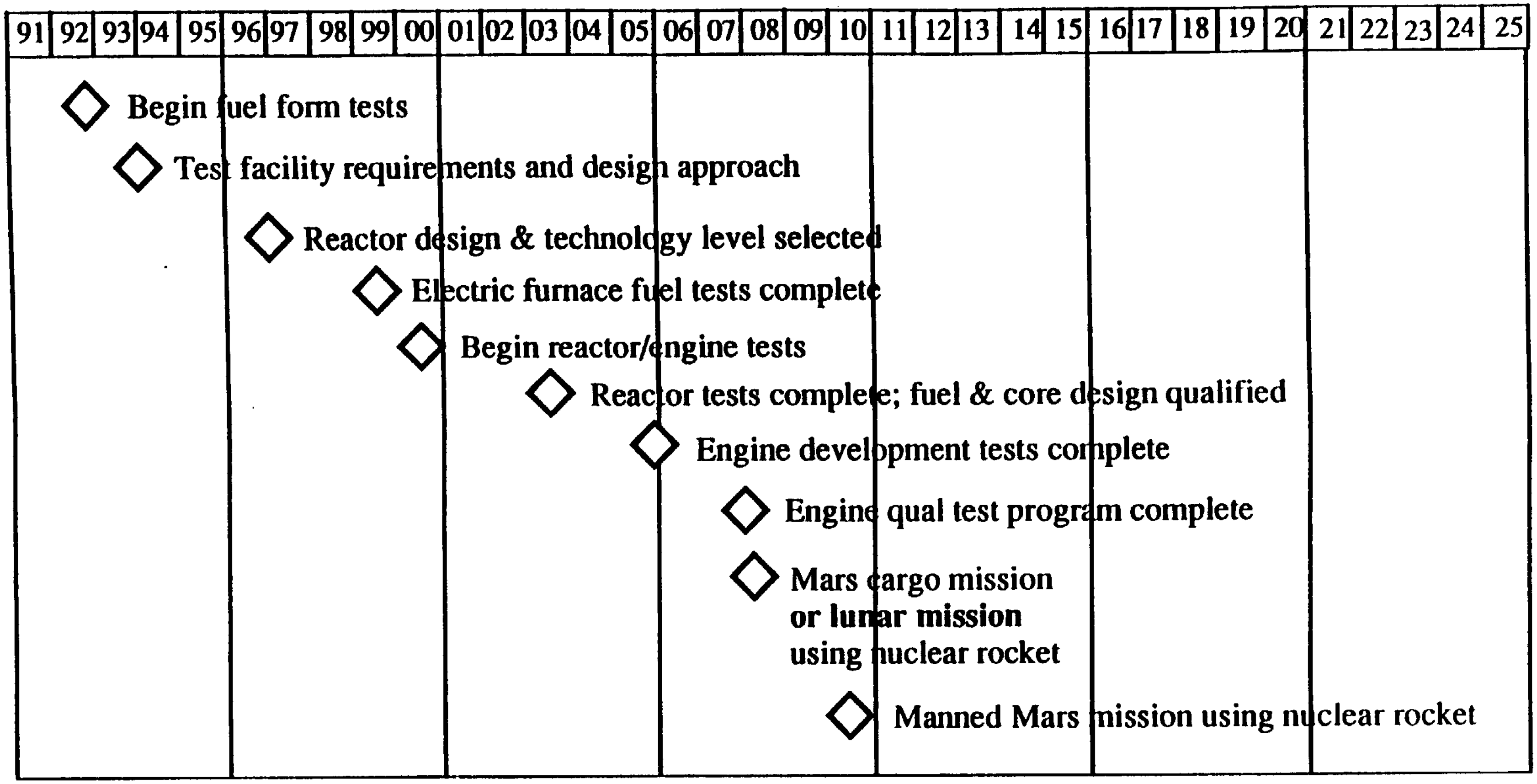
ADVANCED CIVIL SPACE SYSTEMS

NTR Development Program

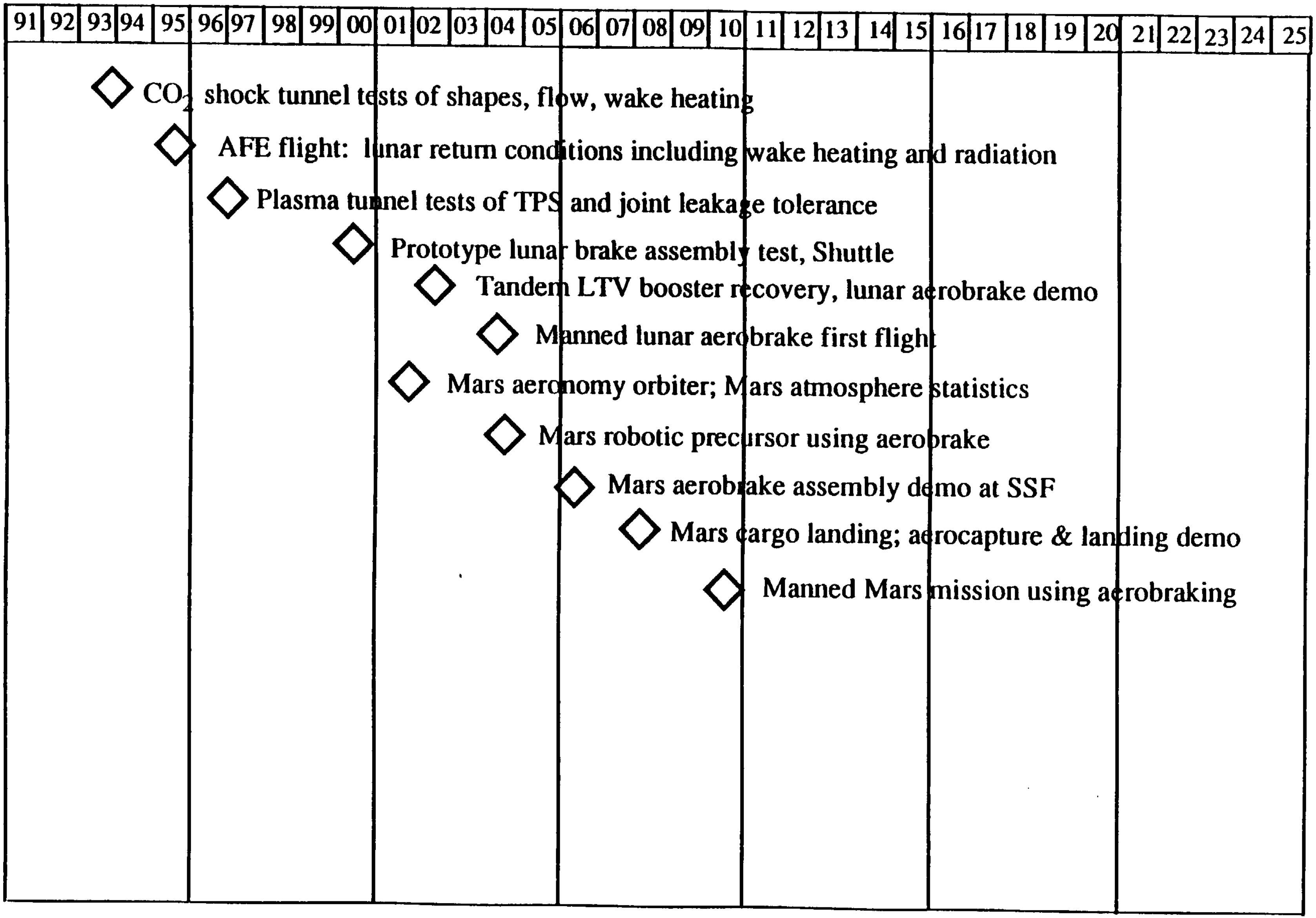
BOEING



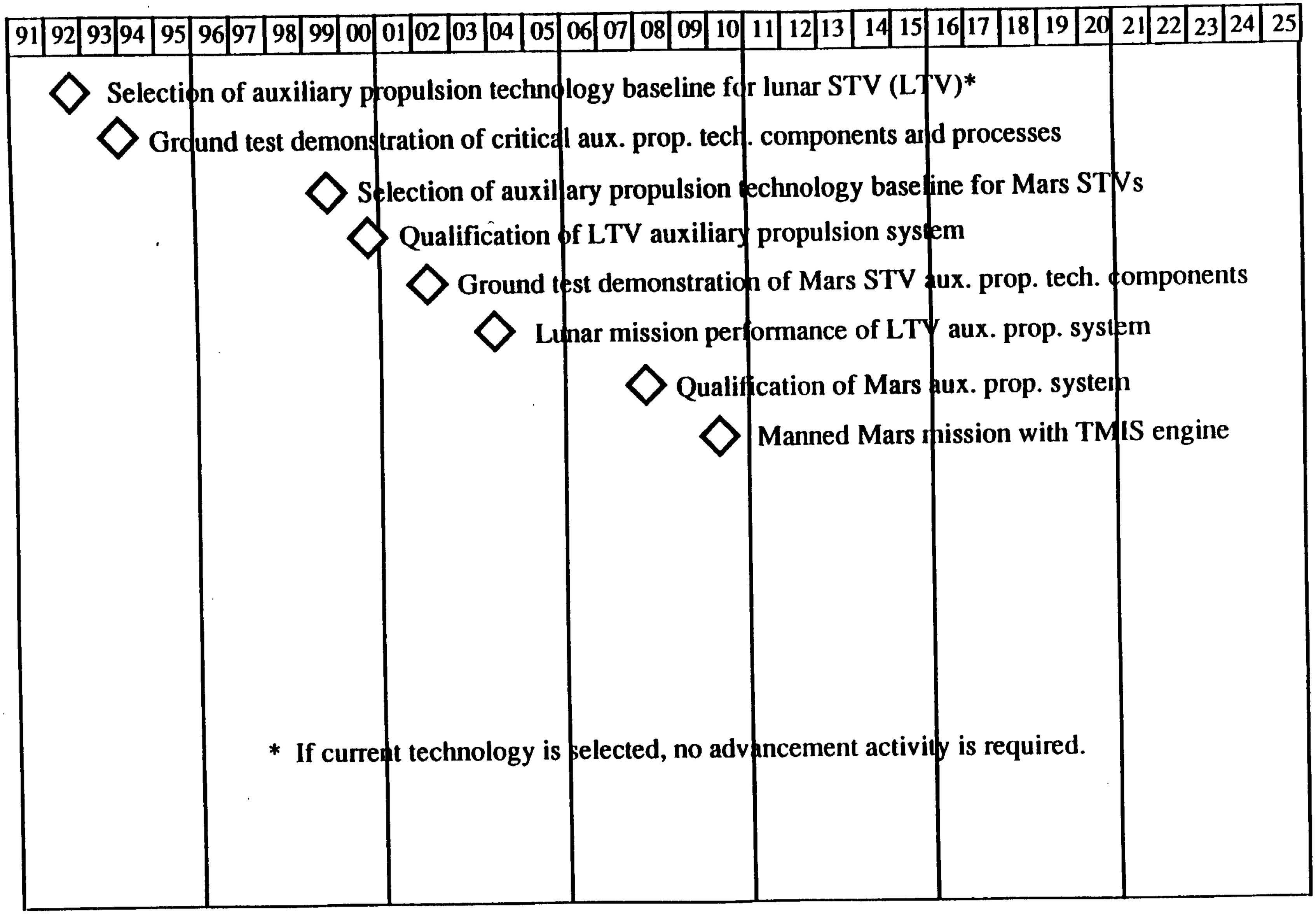
Nuclear Rocket Man-Rating Approach



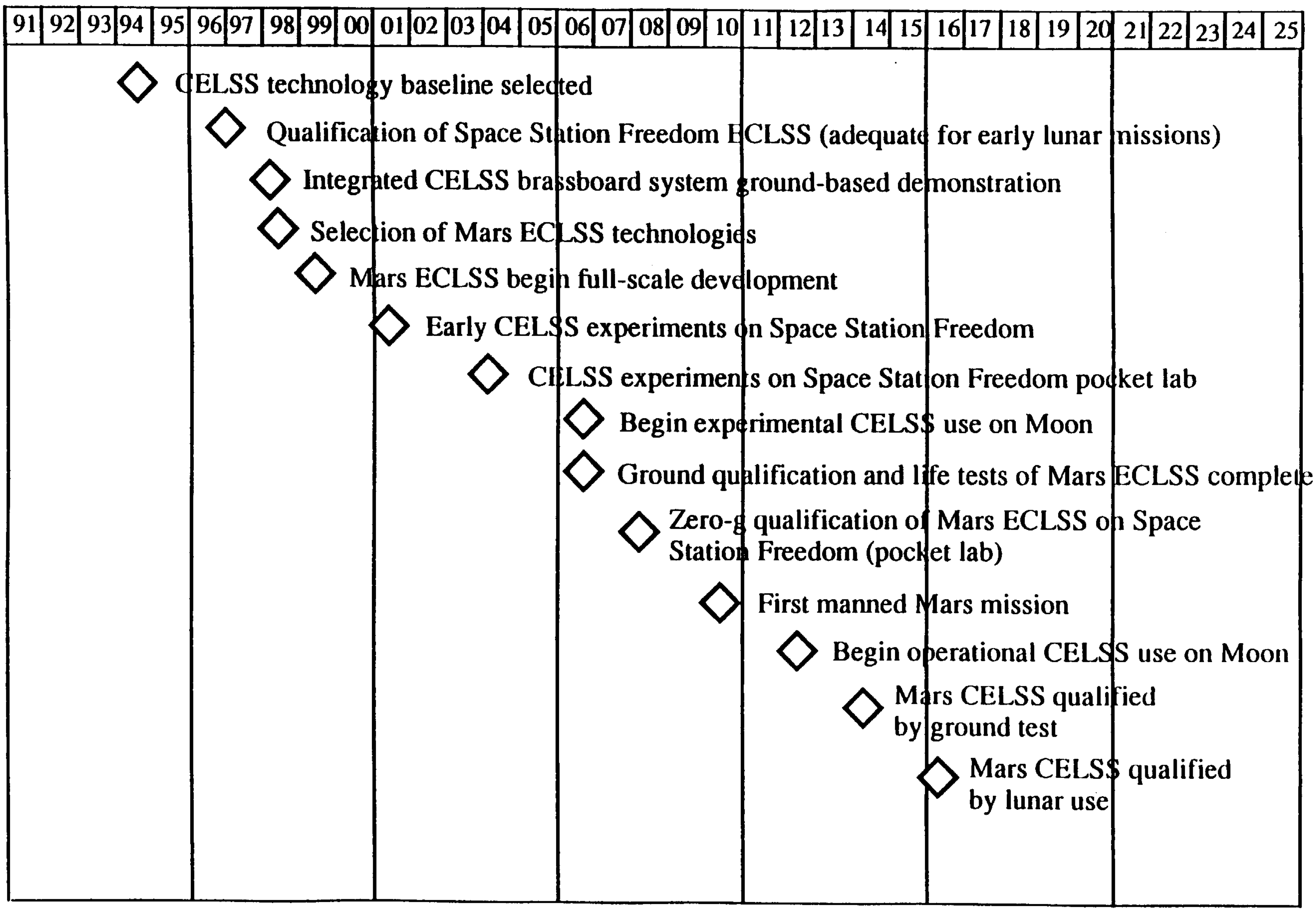
Aerobraking Major Test/Demo Man-Rating Approach



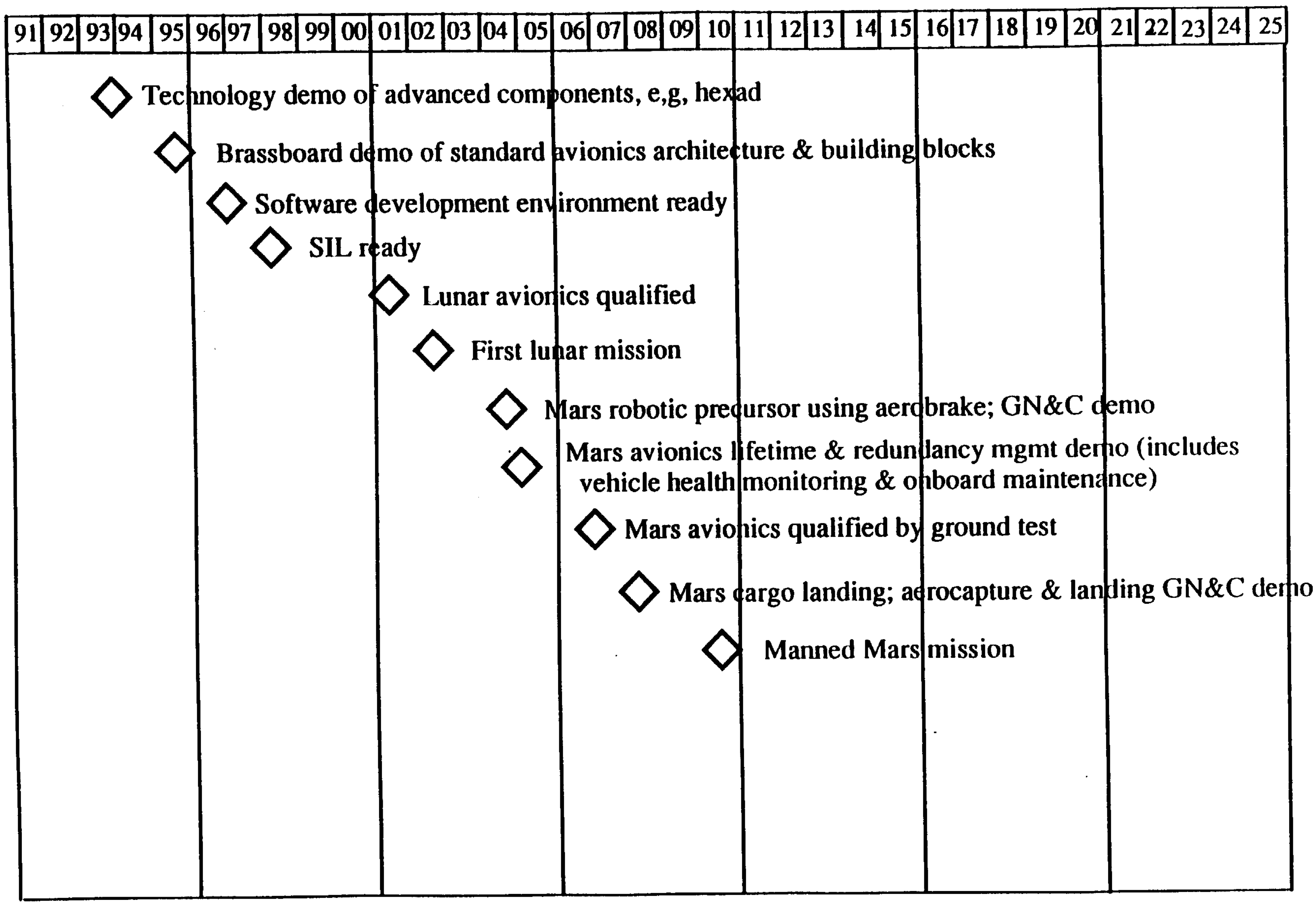
Advanced Auxiliary Propulsion Man-Rating Approach



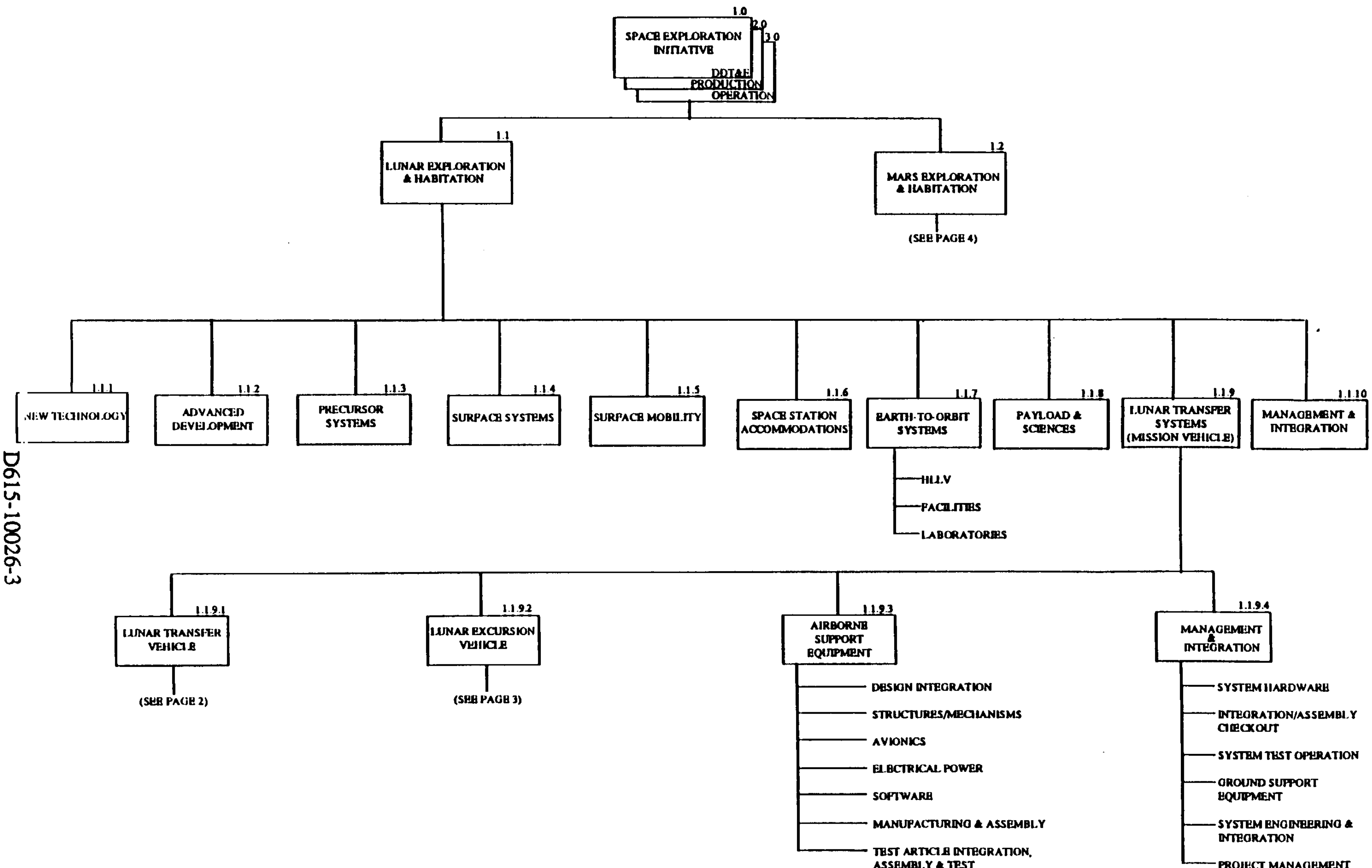
ECLSS Systems Man-Rating Approach



Avionics Major Test/Demo Man-Rating Approach



SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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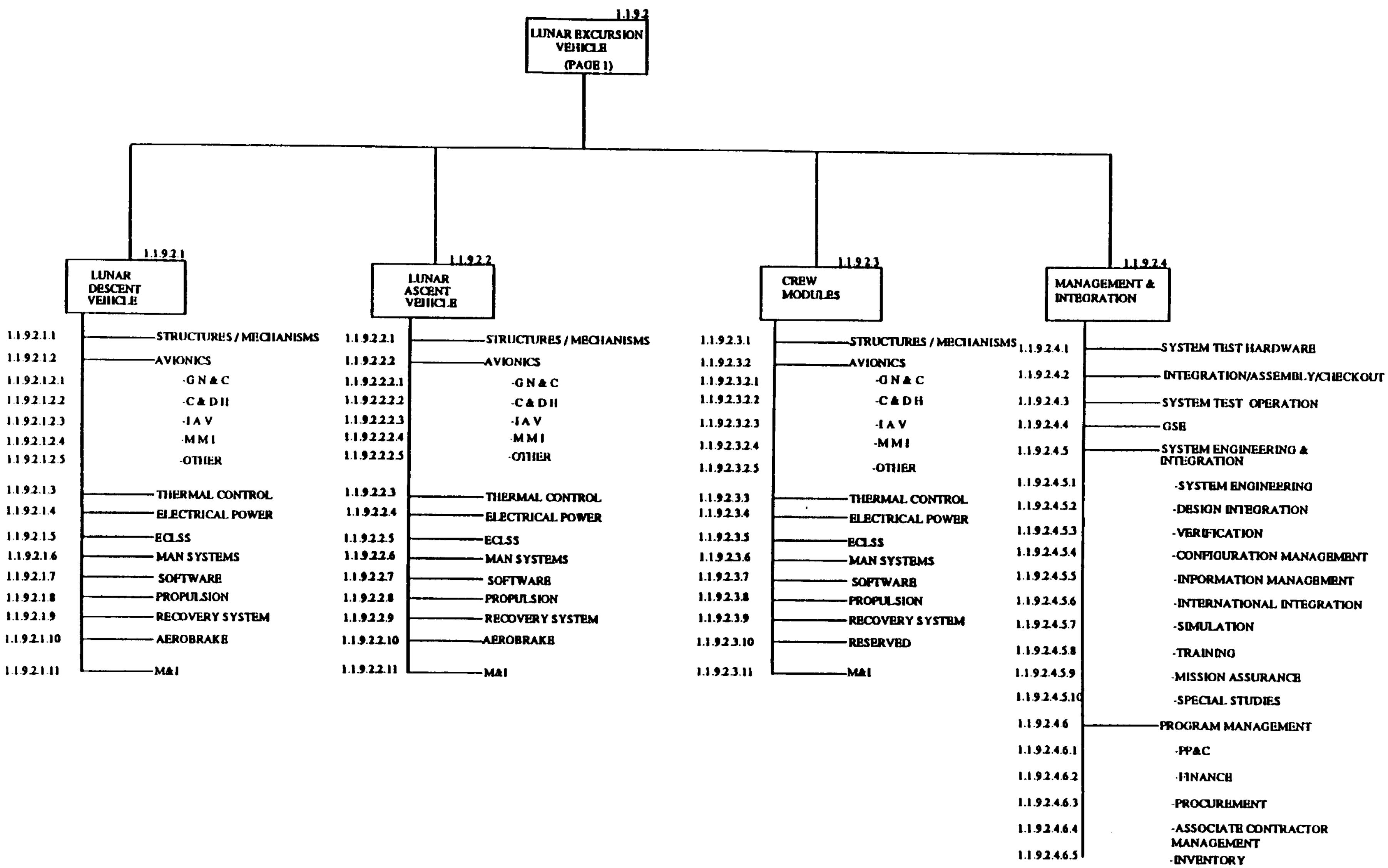
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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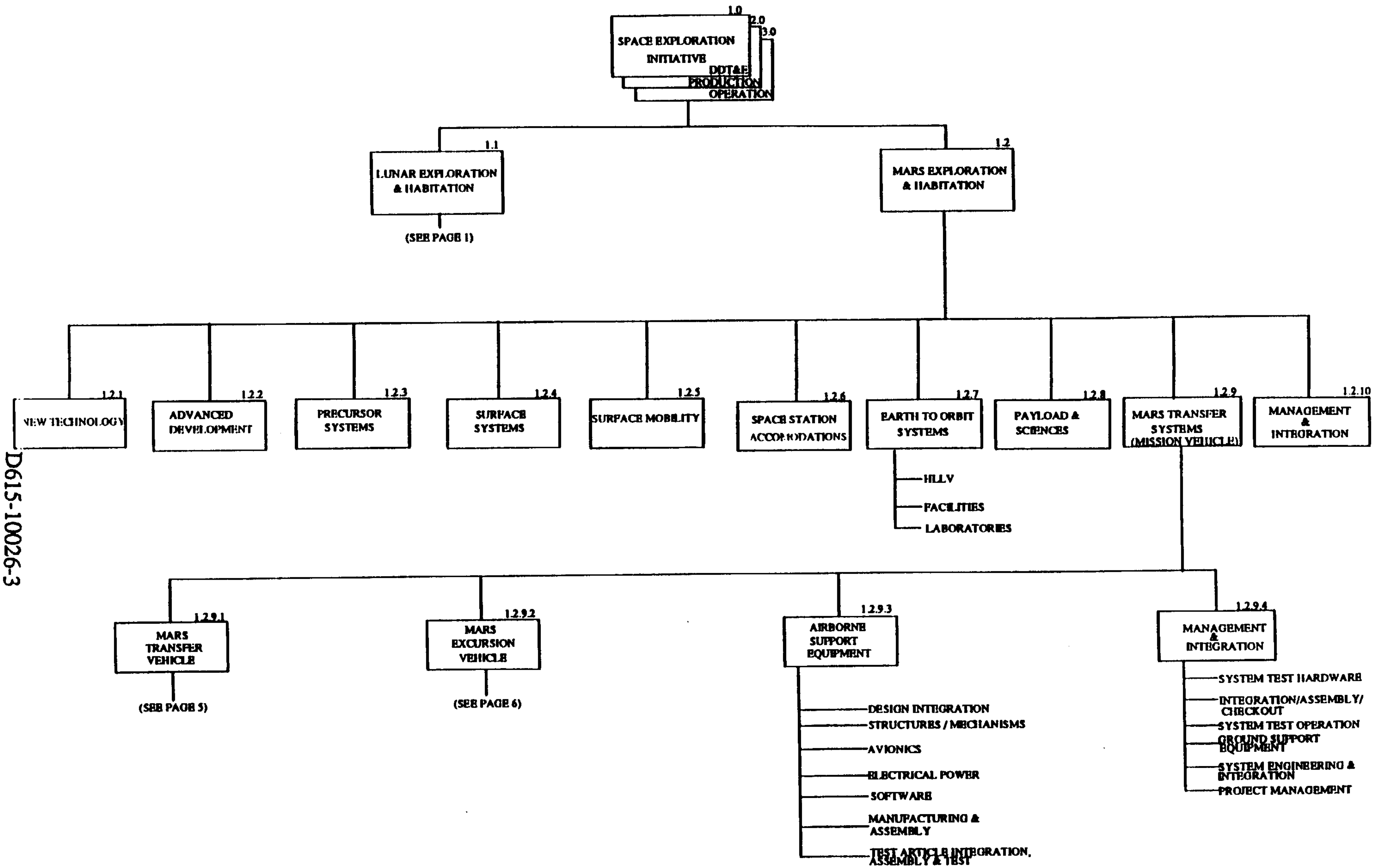
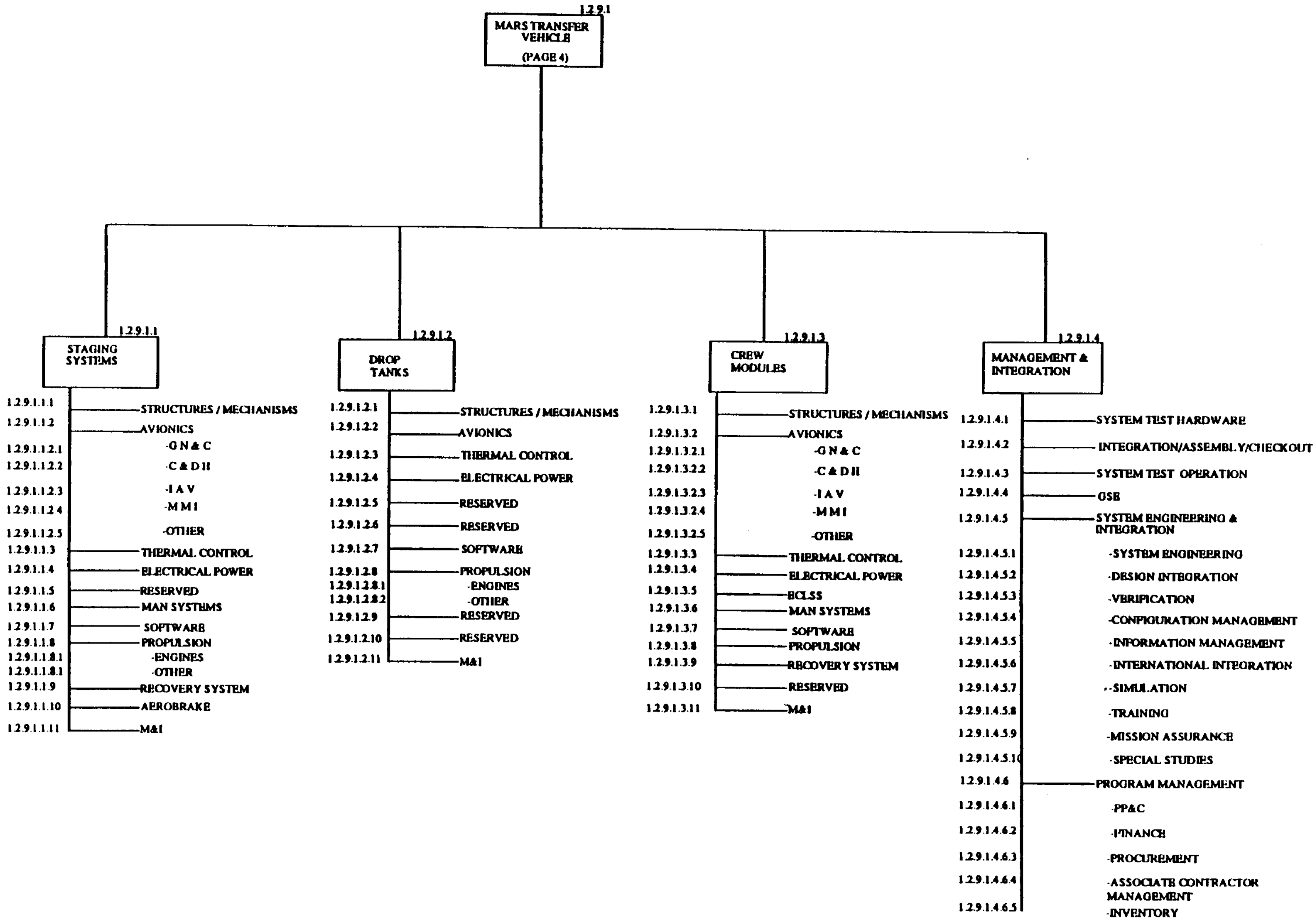
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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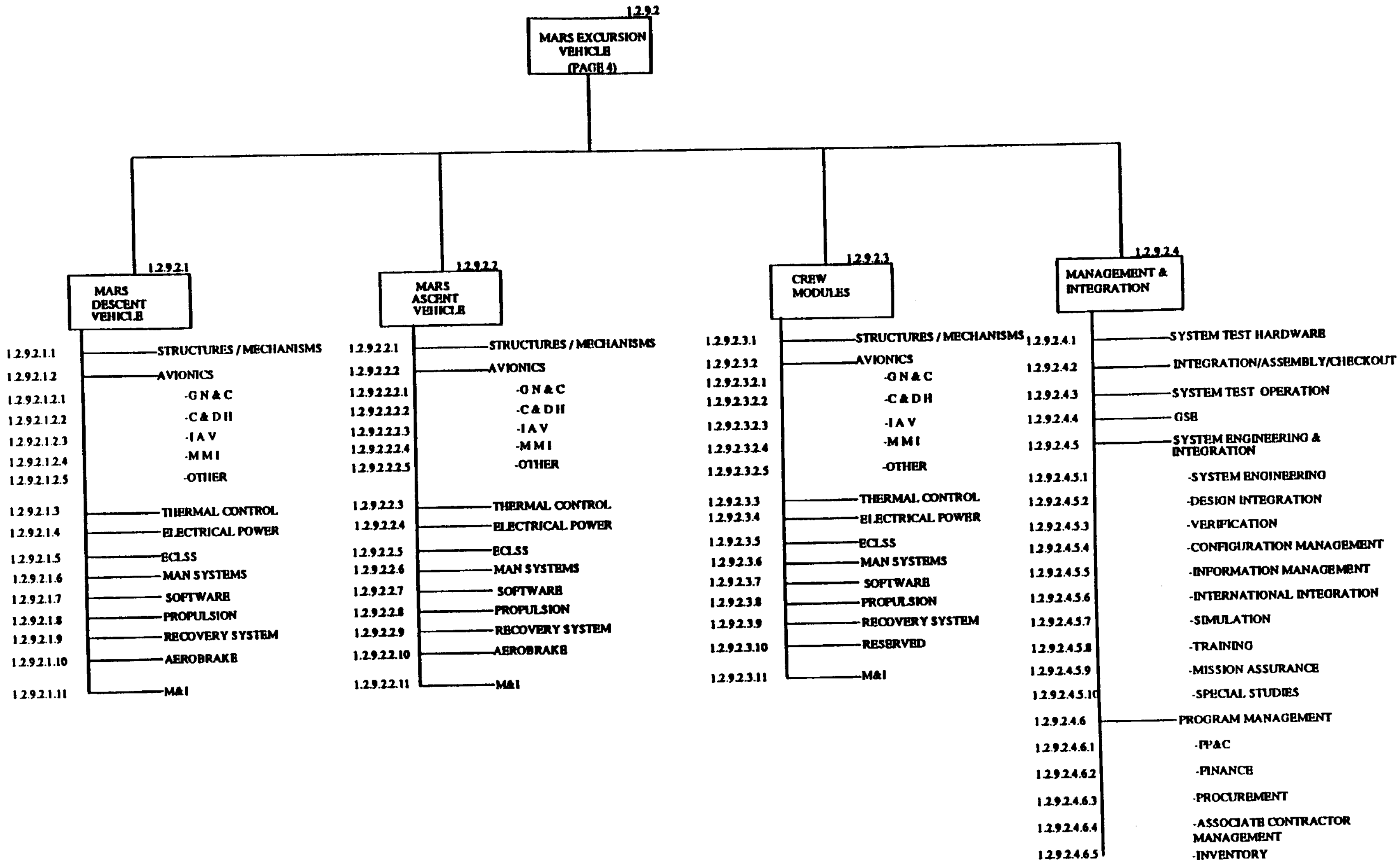
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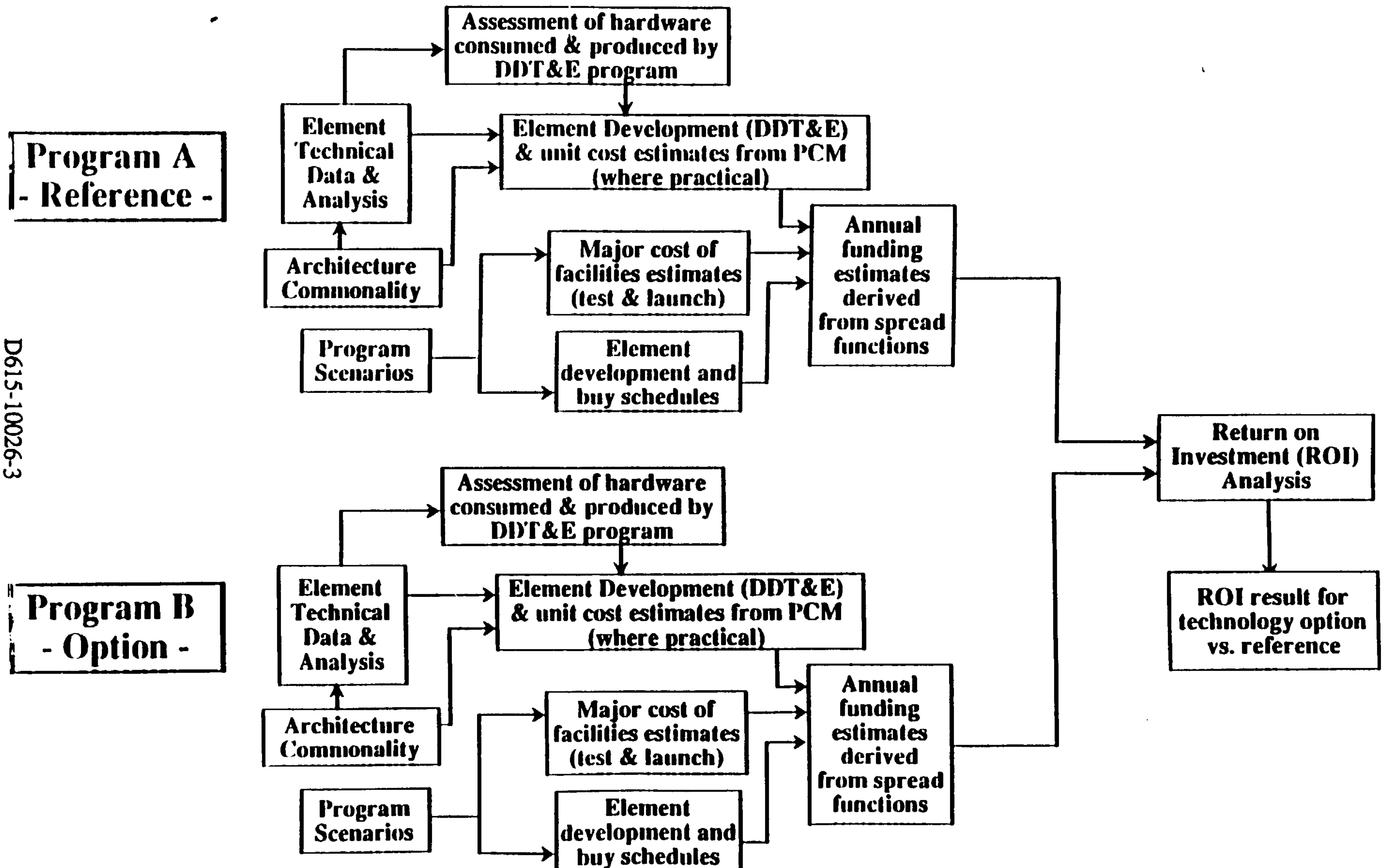
Costing Methodology Flow



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Boeing Parametric Cost Model (PCM)

Features

- Designed specifically for advanced system estimating
- Uses company-wide, uniform computerized data base
- Contains historical data compiled since 1969
- Allows direct input of known costs into the estimate

Main Inputs	Results
• Hardware Characteristics - Category (e.g., primary structure, power conditioning, etc.) - Weight (or Thrust) - Complexity - % Off-the-Shelf - Maturity - Quantity - Manufacturing Learning Curve • Support Cost Factors - Systems Engineering - Management - Operations - Spares	• DDT&E and Manufacturing Estimates - Based on previous Boeing programs - Provides first flight unit costs - Excludes test hardware - Excludes fees • New hardware must be relatable to PCM database to produce reasonable estimate • PCM estimates improve with increasing hardware detail.

LCCM Hardware Assignments

Components		Lunar/Mars		
		Minimum	Full Science	Settle /Ind
HLLV	Cargo Carrier & Core	X	X	X
	STME	X	X	X
	Recov PA Mod	X	X	X
Propulsion	Std Avionics Suite	X	X	X
	Adv Space Engine	X	X	X
	NTR Tanks		X	
	MOC Tank	X		X
	MOC Core	X		X
	NTR Stage		X	
	NTR Engine		X	
	NEP Stage			X
	NEP Engine			X
	TMIS Engine	X		X
	TMIS Tank	X		X
	TMIS Core	X		X
Modules	LEO Tanker	X	X	X
	LTV Hab	X	X	X
	LTV	X	X	X
	LEV	X	X	X
	LEV Crew Module	X	X	X
	MTV	X		X
	MTV Crew Module	X	X	X
	MEV	X	X	X
	RMEV			X
	mini-MEV		X	
	MEV Crew Module	X	X	X
	Lunar Aerobrake	X		
	MTV Aerobrake			
	MEV Aeroshell	X	X	X
	MCRV	X	X	X

ADVANCED CIVIL
SPACE SYSTEMS

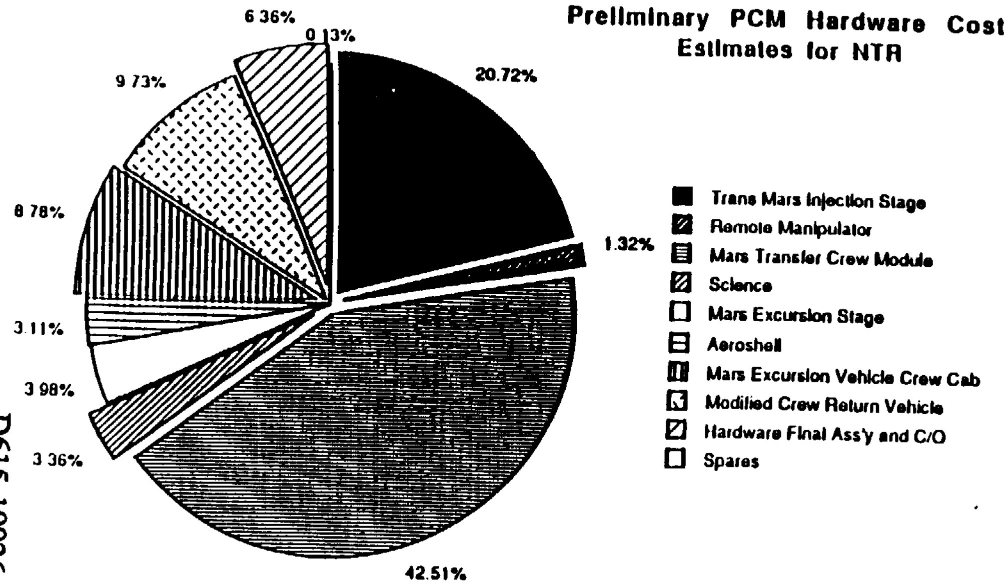
Mars NTR Preliminary PCM Summary

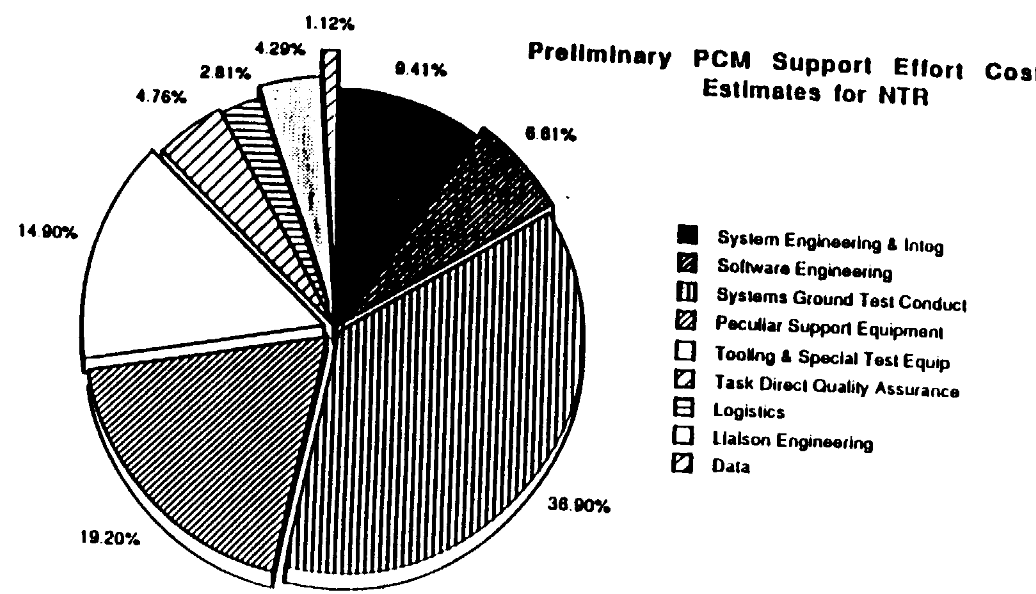
BOEING

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Trans Mars Injection Stage	473.385	532.611	1005.996
Remote Manipulator	19.701	44.303	64.004
Mars Transfer Crew Module	1138.947	925.055	2064.003
Science	100.651	62.517	163.167
Mars Excursion Stage	58.783	134.310	193.093
Aeroshell	99.473	51.556	151.030
Mars Excursion Vehicle Crew Cab	315.766	110.413	426.178
Modified Crew Return Vehicle	273.312	199.326	472.637
Hardware Final Ass'y and C/O	----------	309.013	309.013
Spares	----------	6.180	6.180
Hardware Total Costs	2480.019	2375.283	4855.301
System Engineering & Integration	451.488	----------	451.488
Software Engineering	317.146	----------	317.146
Systems Ground Test Conduct	1771.157	----------	1771.157
Systems Flight Test Conduct	----------	----------	----------
Peculiar Support Equipment	802.895	118.455	921.350
Tooling & Special Test Equipment	----------	715.107	715.107
Task Direct Quality Assurance	----------	228.593	228.593
Logistics	135.055	----------	135.055
Liaison Engineering	205.851	----------	205.851
Data	53.665	----------	53.665
Training	O/H	----------	----------
Facilities Engineering	O/H	----------	----------
Safety	O/H	----------	----------
Graphics	O/H	----------	----------
Outplant	O/H	----------	----------
Program Management	O/H	----------	----------
Support Effort Total	3737.247	1062.155	4799.402
Total Estimate	**6217.266**	**3437.438**	**9654.703**

O/H = Overhead charge (included in above costs)

Mars NTR Preliminary PCM Summary - continued





	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	TMIS Core	439.2	0	2.79	1225.368	225	3.5	
3	TMI Tank	32		2.79	89.28	48.9	2	
4	TMIS Tank/MOC	17	0	2.79	47.43	8.63	2	
5	TMIS Nuc Eng	468	0	2.79	1305.72	150	2	
6	Test Facil	0	2255	1	2255	0	1	
7	Crew Module	1113	0	2.79	3105.27	1281	3.1	
8	MEV Stg	66.8		2.79	186.372	107.5	3.5	
9	MEV Engine	0	0	2.79	0	8	5	
10	MEV Aeroshell	112.1		2.79	312.759	64.7	2	
11	Mev CM	142		2.79	396.18	126.5	3.5	
12	ECCV	280		2.79	781.2	214	3.5	
13								

	I	J	K	L	M	N	O
1	DDT&E no Fe	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/	Fee
2	2012.868	8	2173.89744	0.3	67.5	72.9	
3	187.08	8	202.0464	2	97.8	105.624	
4	64.69	8	69.8652	2	17.26	18.6408	
5	1605.72	8	1734.1776	2	300	324	
6	2255	8	2435.4	0	0	0	
7	7076.37	8	7642.4796	0.25	320.25	345.87	
8	562.622	8	607.63176	1	107.5	116.1	
9	40	8	43.2	7	56	60.48	
10	442.159	8	477.53172	1	64.7	69.876	
11	838.93	8	906.0444	1	126.5	136.62	
12	1530.2	8	1652.616	1	214	231.12	
13		Grand Total	17742.8437				1375.6068

LCC Cost Buildup Example From PCM
- NTR -

	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	TMIS Core	439.2	0	2.79	1225.368	225	3.5	
3	TMI Tank	32		2.79	89.28	48.9	2	
4	TMIS Tank	17	0	2.79	47.43	8.63	2	
5	TMIS Nuc Eng	468	0	2.79	1305.72	150	2	
6	Test Facil	0	2255	1	2255	0	1	
7	Crew Module	1113	0	2.79	3105.27	1281	3.1	
8	MEV Slg	66.8		2.79	186.372	107.5	3.5	
9	MEV Engine	0	0	2.79	0	8	5	
10	MEV Aeroshell	112.1		2.79	312.759	64.7	2	
11	Mev CM	142		2.79	396.18	126.5	3.5	
12	EOCV	280		2.79	781.2	214	3.5	
13								

	I	J	K	L	M	N	O
1	DDT&E no Fe	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/	Fee
2	2012.868	8	2173.89744	0.3	67.5	72.9	
3	187.08	8	202.0464	2	97.8	105.624	
4	64.69	8	69.8652	2	17.26	18.6408	
5	1605.72	8	1734.1776	2	300	324	
6	2255	8	2435.4	0	0	0	
7	7076.37	8	7642.4796	0.25	320.25	345.87	
8	562.622	8	607.63176	1	107.5	116.1	
9	40	8	43.2	7	56	60.48	
10	442.159	8	477.53172	1	64.7	69.876	
11	838.93	8	906.0444	1	126.5	136.62	
12	1530.2	8	1652.616	1	214	231.12	
13		Grand Total	17742.8437				1375.6068

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Development Risk Assessment For Aerobraking By Function

MISSION FUNCTION	BRAKE SIZE	ATMOSPHERE KNOWLEDGE & UNCERTAINTY	TARGET FOR ENTRY: GN&C PRECISION	HEATING/TPS	AERO PASS GN&C PRECISION REQUIRED
Lunar return Earth landing	Small, no ass'y required	Accurate knowledge, low uncert. effect	Very high	State-of-the-Art	State-of-the-Art
Lunar return Earth landing	Moderate requires assembly	Accurate knowledge, high uncert. effect	Very high	State-of-the-Art	Believed State-of-the-Art
Mars landing from orbit	Large, requires assembly	Poor knowledge, low uncert. effect	Can be high, e.g. done from Mars orbit	State-of-the-Art	Believed State-of-the-Art
Mars return Earth landing	Small, no ass'y required	Accurate knowledge, moderate uncertainty effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Accurate knowledge, high uncert. effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Poor knowledge, high uncert. effect	Poor, unless nav-aids in Mars orbit	High heating rates, some TPS advancement needed	Advancements required

005/EAW/Disk 1
1/21/91